Journal

of

THE ROYAL AERONAUTICAL SOCIETY

SEPTEMBER 1959

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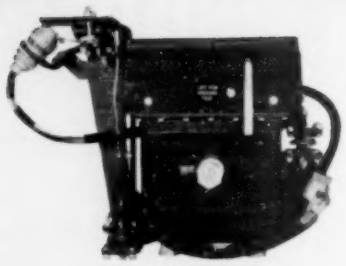

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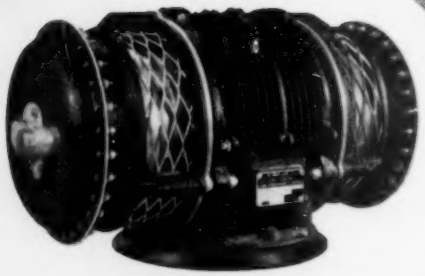

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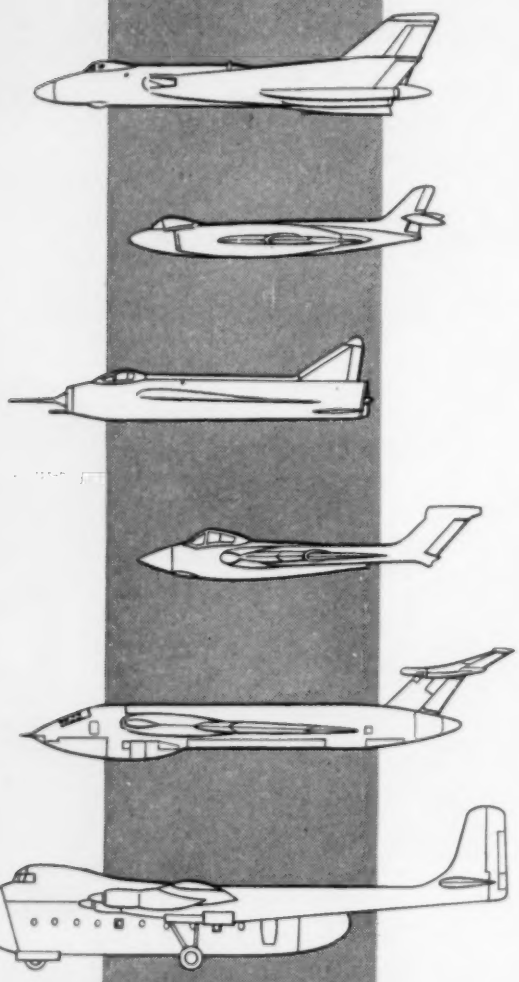

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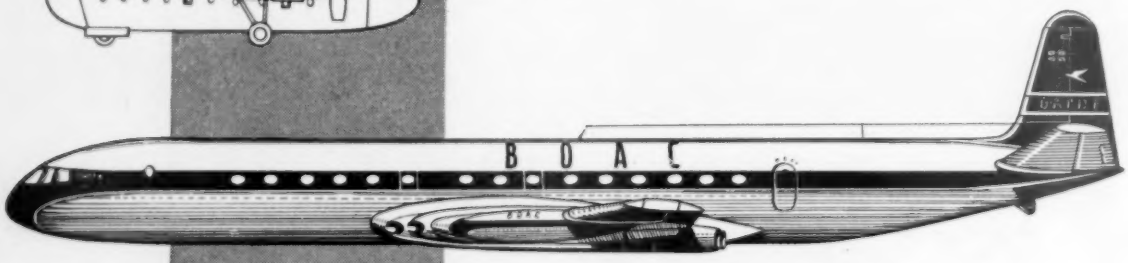

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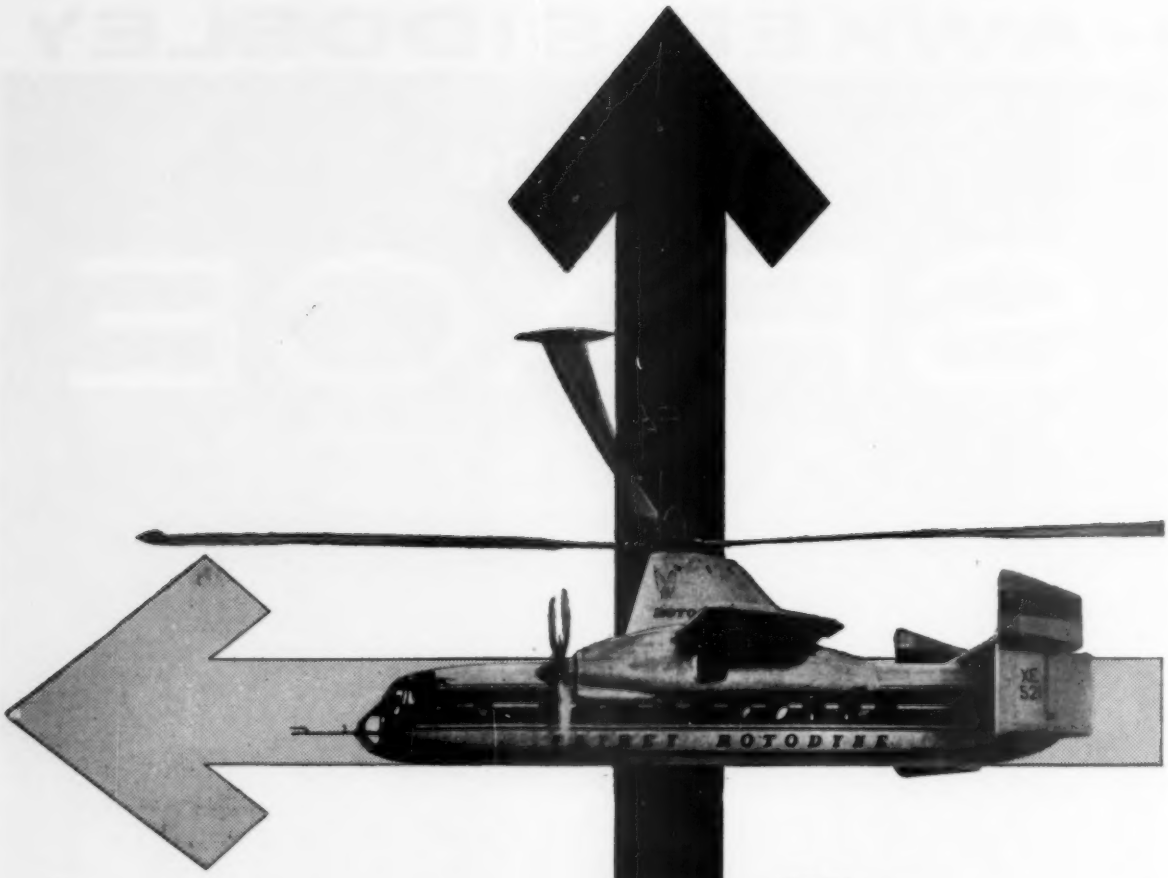

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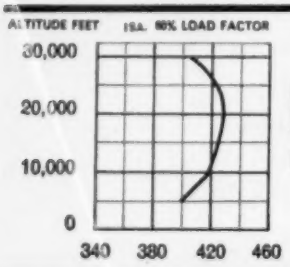

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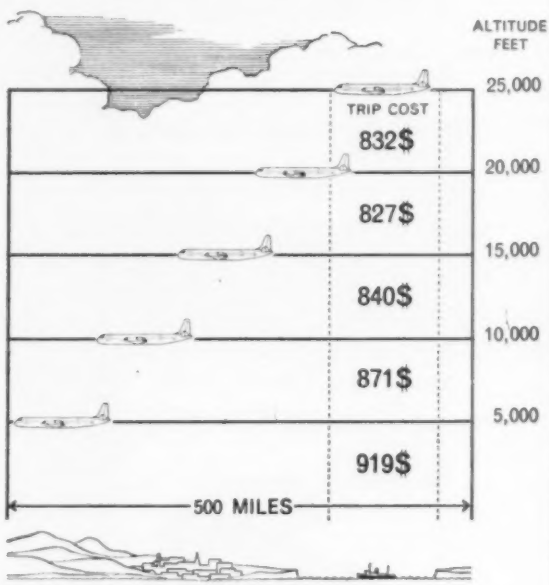

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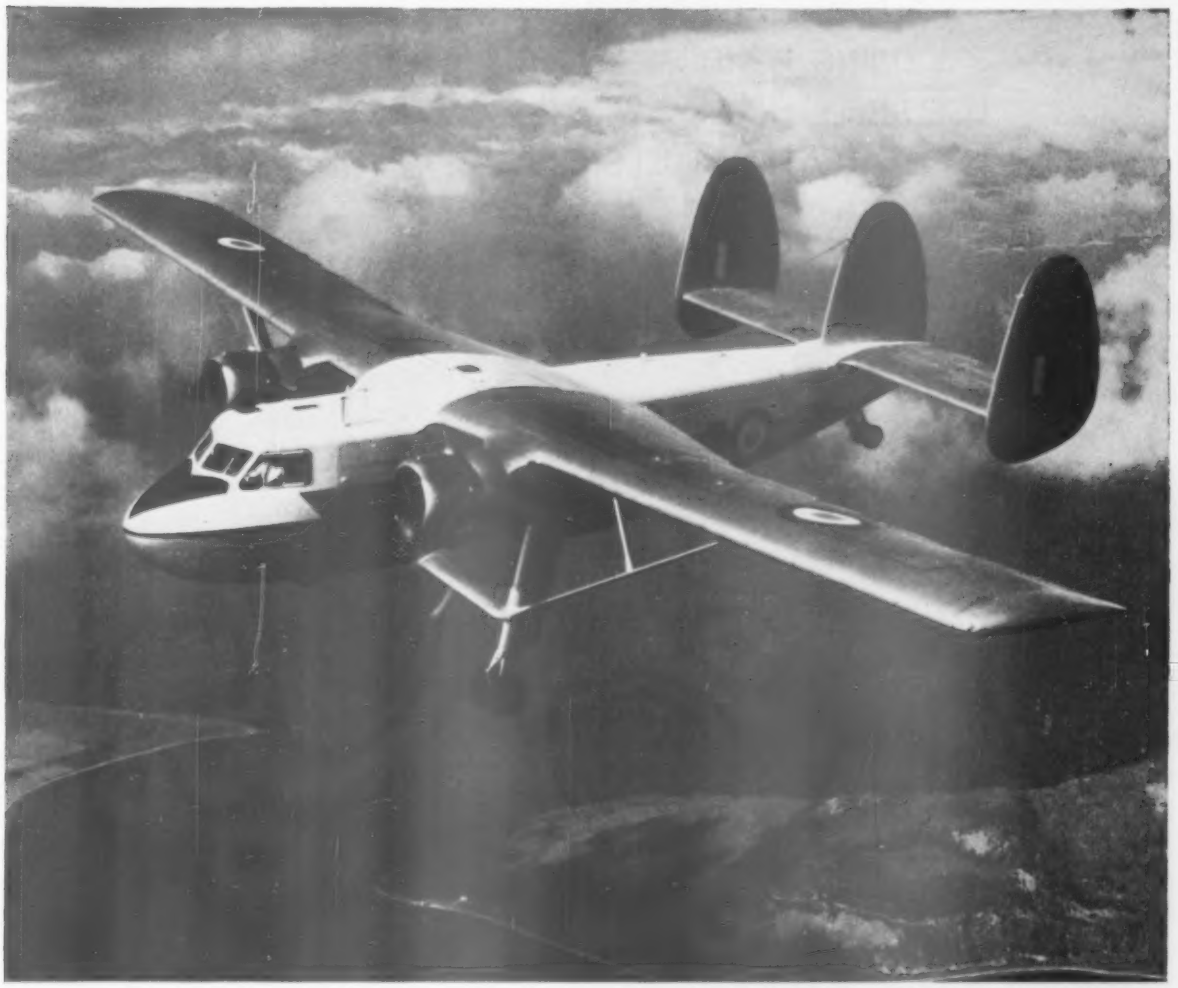

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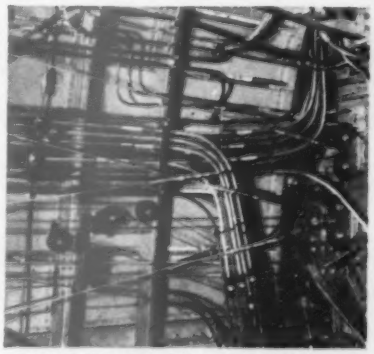

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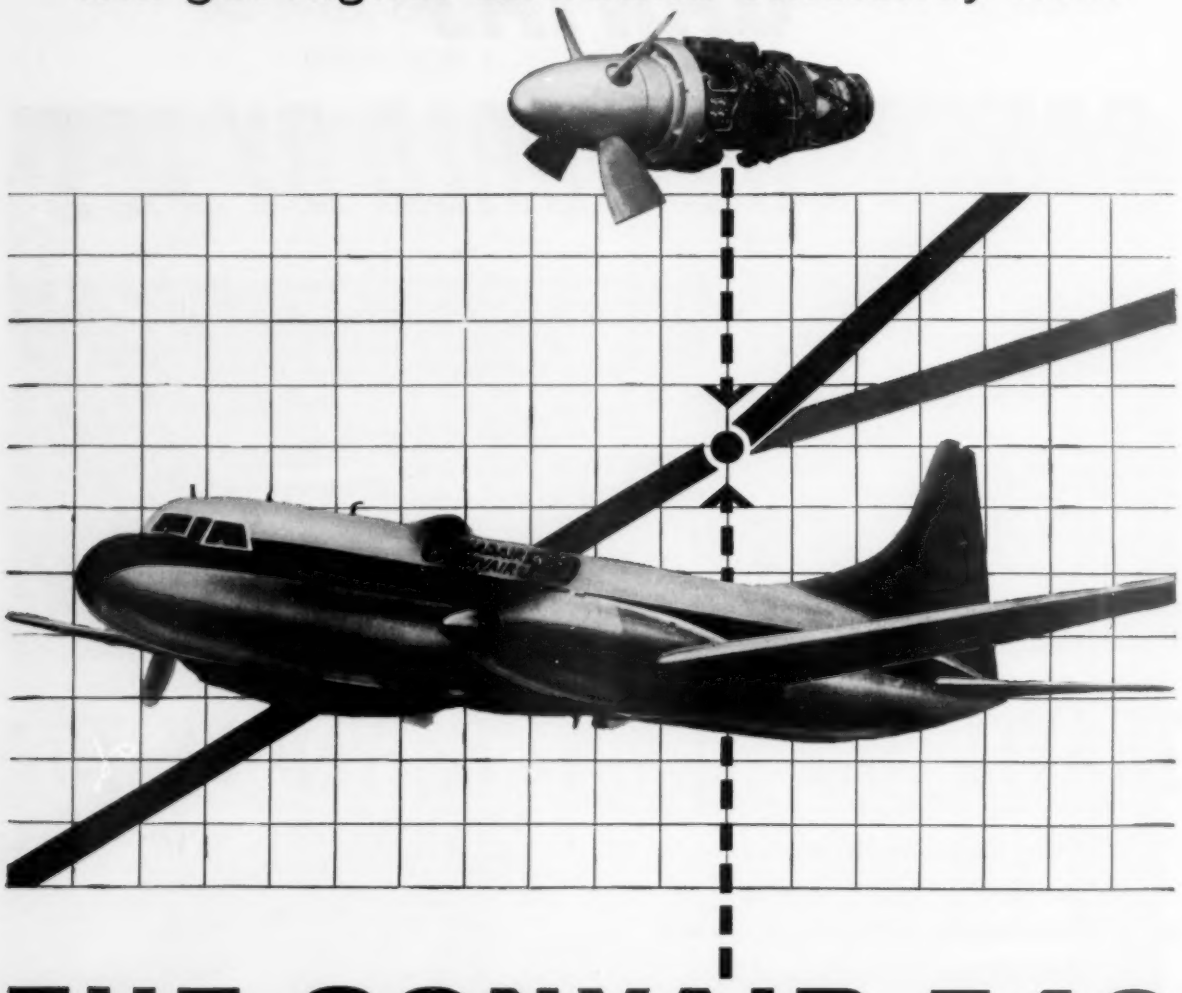

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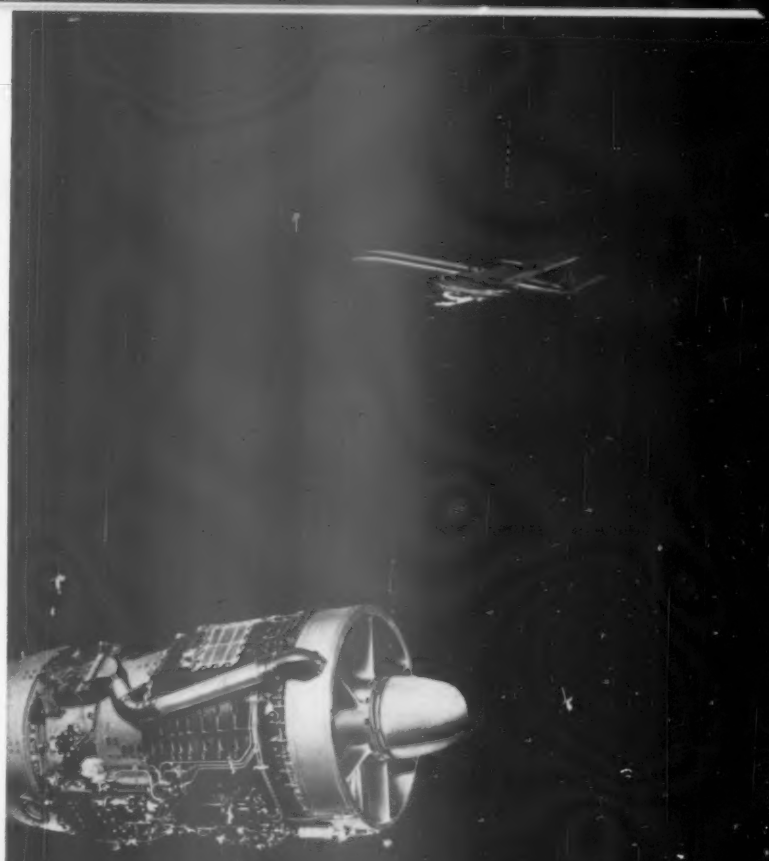

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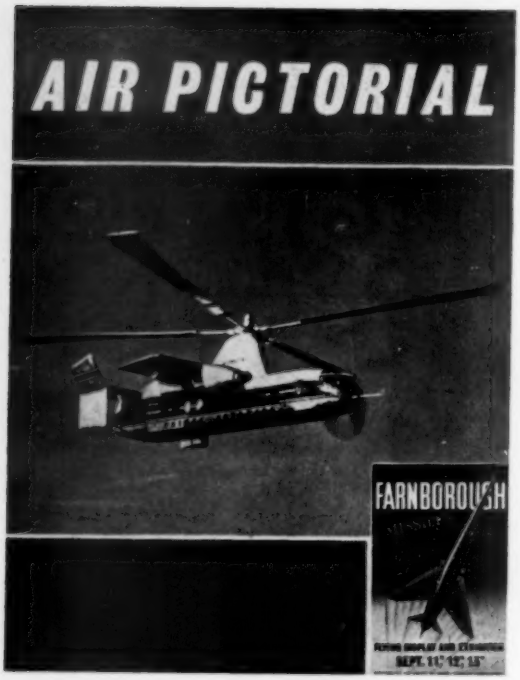

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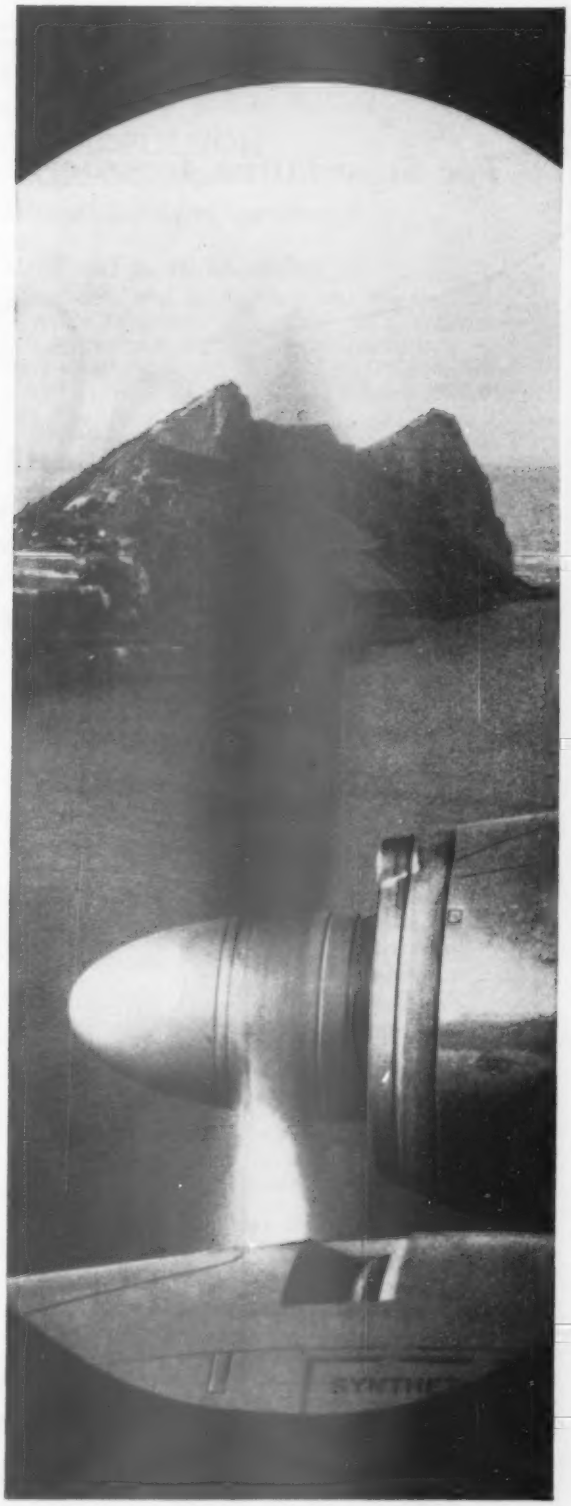

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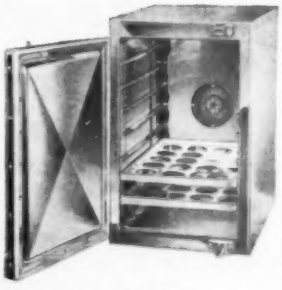

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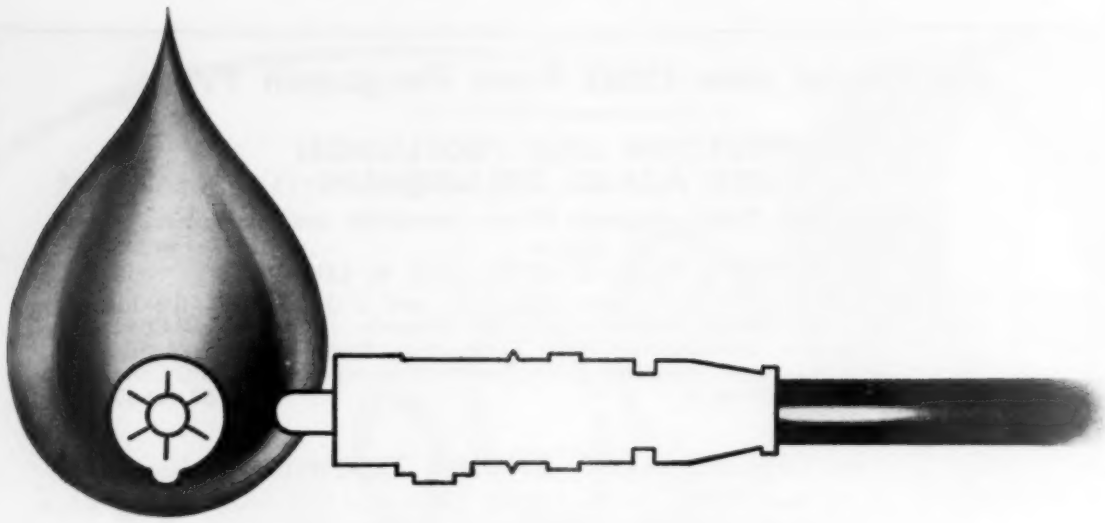

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All communications for publication in the JOURNAL *should be addressed to the Editor,* JOAN BRADBROOKE, *A.R.Ae.S.; all communications on general matters affecting the Society should be addressed to the Secretary,* A. M. BALLANTYNE, *T.D., B.Sc., Ph.D., Hon.F.C.A.I., F.I.A.S., F.R.Ae.S.*

None of the papers or paragraphs must be taken as expressing the opinion of the Council of the Royal Aeronautical Society unless otherwise stated.

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NOTICES

Special General Meeting—29th October
The Society and The Helicopter Association

Members are reminded that a Special General Meeting of the Voters of the Society will be held on Thursday 29th October at 6.30 p.m. at the offices of the Society. Particulars of the Meeting and the Resolution which is to be voted upon were published in the August JOURNAL, p. XXXVII, and have been sent by air mail to overseas members.

DIARY

LONDON

4th September

SYMPOSIUM—Operational Problems of Take-off and Landing. L. P. Coombes, E. W. Pike, Captain B. O. Prowse, E. S. Calvert. Library, 4 Hamilton Place. 7 p.m.

30th October

INAUGURAL LECTURE—MAN POWERED AIRCRAFT GROUP—Resumé of the Theory and Practice. H. B. Irving, Dr. D. R. Wilkie, T. R. F. Nonweiler, B. S. Shenstone. Library, 4 Hamilton Place. 7 p.m.

17th November

LECTURE—Magneto-Gasdynamics. J. A. Shercliff. Library, 4 Hamilton Place. 7 p.m.

19th November

FIFTEENTH BRITISH COMMONWEALTH LECTURE—Some Recent Progress in Air Survey with Particular Reference to Newly Developed Territories. W. P. Smith. The Institution of Mechanical Engineers, 1 Birdcage Walk, S.W.1. 6 p.m. (Tea at 5.30 p.m.)

ASTRONAUTICS AND GUIDED FLIGHT SECTION

22nd October

LECTURE—Problems of Interplanetary Navigation. T. R. F. Nonweiler. Church House, Westminster, S.W.1. 7 p.m.

3rd November

LECTURE—Control Mechanisms Dr. F. Errington. Church House, Westminster, S.W.1. 7 p.m.

GRADUATES' AND STUDENTS' SECTION

23rd September

Film Show. Library, 4 Hamilton Place. 7.30 p.m.

7th October

Lunar Probes. J. E. Allen. Library, 4 Hamilton Place. 7.30 p.m.

28th October

The Setting of Aircraft Specifications. R. H. Whitby. Library, 4 Hamilton Place. 7.30 p.m.

BRANCHES

29th September

Luton—Ballistic Missile Design. E. C. Cornford. (Joint lecture with E.E.C.). English Electric Co., Stevenage. 6.30 p.m.

1st October

Isle of Wight—Aircraft Accident Investigation. R. C. Clarke. Clubhouse, Saunders-Roe Sports and Social Club, Church Path, E. Cowes. 6 p.m.

13th October

Luton—Instruments and Flight Control Systems for Transport Aircraft. W. H. McKinley. Napier Senior Staff Canteen, Luton Airport. 6.15 p.m.

17th October

Luton—Visit to Shuttleworth Collection, Old Warden.

22nd October

Isle of Wight—Air Traffic Control and Jet Operations. H. E. Smith. Clubhouse, Saunders-Roe Sports and Social Club, Church Path, E. Cowes. 6 p.m.

Seventeenth Brancker Memorial Lecture

The Seventeenth Brancker Memorial Lecture of the Institute of Transport will be given by Captain E. V. Rickenbacker, Chairman of Eastern Air Lines Inc. on "World Peace Through Air Transport" at 5.45 p.m. at 66 Portland Place, W.1. The meeting is open to visitors.

Anniversary Lunch—12th January 1960

The Prime Minister has accepted an invitation to be the Guest of Honour and the principal speaker at an Anniversary Lunch which is to be held in London on 12th January 1960 to celebrate both the 94th anniversary of the Society and the Golden Jubilee of so much that happened in British Aviation in 1910.

The Council feel that a mid-day occasion, rather than a dinner or reception in the evening, would enable more members and their guests to attend for there would be no necessity for over-night accommodation in London.

Further particulars will be published shortly. Please PUT THIS DATE ON YOUR CALENDAR.

Honours Awarded to Members

Birthday Honours

The following Members of the Society were included in the Birthday Honours List:—

C.B.

Rear Admiral A. J. Tyndale-Biscoe (*Fellow*)

O.B.E.

E. J. Dickie (*Associate*)
K. O. Grant (*Associate Fellow*)
H. B. Irving (*Fellow*)
W. J. A. Thorn (*Associate Fellow*)

M.B.E.

D. Cameron-Douglas (*Associate Fellow*)

Honorary Degree

Dr. G. V. Lachmann (*Fellow*) has been awarded the Honorary Degree of Doctor of Engineering (Dr.-Ing.e.H.), by the Technical University of Aachen.

Elections

The following is a list of new members and transfers of membership of the Society:—

Associate Fellows

John Gerald Thomas Dawson Atkinson
Albon Beedle (*from Associate*)
Arthur Robert Chandler (*from Graduate*)
Percy George Chaplin (*from Associate*)
Richard Albert Fuller (*from Associate*)
Gilbert Ratcliffe Hadfield (*from Associate*)
Arthur William Hallatt
Keith Gordon Hodson (*from Graduate*)
Walter Raymond Keeble
Ian McLaren (*from Associate*)
Gordon Lorimer McWilliam
Philip Holmes O'Neill
Jack Shallett Palmer
Barry Arthur Phillips
Eugene Lewis Place
Donald Rowley
John Howard Sargeant (*from Graduate*)
Sheikh Siraj (*from Graduate*)
Richard Stanton-Jones (*from Graduate*)
Arthur Charles Whitfield

Associates

William James Brown
John Henry Cadiz
Peter John Carnell Chapman
Albert Edward Clifford (*ex Associate*)
Robert Ernest Davison
Ian Hamilton Kemp
Norman Frank Laurence
Cyril Spear

Graduates

William Hobart Atwood
David Anthony Brooks
Paul Victor Cope (*from Student*)
David Ian Cowie
Reginald Kenneth Foster
James Brian Hargrave
Alan Jennings
Rajendra Nath
Brian Smith
Gordon John Nicoll Smith

Students

Daljit Singh Badwal
Frank James Couch
Anthony Edward Cutler
Anthony Sydney De'Forge Dedman
John Torry McMillan
Barry Willstrop

Companion

Thomas Harris Atkinson

ROYAL AERONAUTICAL SOCIETY SCHOLARSHIPS AND GRANTS

The following Scholarship and Grant have been awarded by the Council for 1959-60:—

The Geoffrey de Havilland Memorial Scholarship—Awarded for a second year to Mr. A. J. W. Smith of 352 Woodham Lane, New Haw, Weybridge, Surrey. He will continue his post-graduate work on Aerodynamics and Aircraft Design at the College of Aeronautics, Cranfield.

An Educational Grant has been awarded to Mr. J. D. Watkins of 113 Howlands, Welwyn Garden City, Herts., to enable him to take the two year Diploma Course at the College of Aeronautics, Cranfield. Mr. Watkins is 27 years old, has passed the Associate Fellowship Examination and wishes to specialise in Aircraft Design and Aircraft Production.

(The Charter Scholarship and the Edward Busk Studentship in Aeronautics were not awarded this year.)

REPRESENTATIVES ON OTHER BODIES

Mr. R. M. Cracknell (*Associate Fellow*) of de Havilland Propellers Ltd., has been appointed the Society's Representative on the British Standards Institution Technical Committee MEE/143/Gas Turbines in succession to Mr. F. M. Owner.

S.B.A.C. UNIVERSITY SCHOLARSHIPS

B. Maskew (*Student*) and B. R. Perkin (*Student*), both of Blackburn Aircraft Ltd., are among the seven successful candidates to be awarded University Scholarships for 1959 by the Society of British Aircraft Constructors Ltd. Both Mr. Maskew and Mr. Perkin will go to the College of Aeronautics.

NEWS OF MEMBERS

A. G. J. BAKER (*Graduate*) has been awarded a Diploma by the College of Aeronautics for the two-year post-graduate Diploma course and has also been awarded the Governors' Prize and the S.B.A.C. Prize in Aircraft Propulsion.

CAPT. E. C. BEARD (*Associate Fellow*), formerly Assistant Director, R.N.I, M.O.S., is now Director of Royal Naval Aircraft Research and Development.

SQN. LDR. E. N. BESWICK (*Associate Fellow*), formerly at R.A.F., Nicosia, has been posted to No. 27 Maintenance Unit, Shawbury, as Senior Technical Officer.

D. S. BUCKLE (*Student*), formerly with the Fleet Air Arm is now a Technical Assistant in the Project Engineering Department, Air Trainers Link Ltd., Aylesbury.

D. CAMERON (*Fellow*), formerly Director of Atomic Weapons (Plans), Ministry of Supply, is now Chief Superintendent, A. and A.E.E., Boscombe Down.

ARTHUR EVANS (*Associate Fellow*), formerly an Aircraft Designer, Avro Aircraft Ltd., Toronto, is now a Senior Design Engineer, Lockheed Aircraft Corp., Georgia.

J. R. FINNIMORE (*Associate Fellow*), formerly Design Superintendent, B.O.A.C., has been appointed Aircraft Development Manager.

B. S. GROVER (*Graduate*), formerly a Senior Stressman, Auster Aircraft Ltd., is now an Aeronautical Engineer, Hindustan Aircraft Ltd., Bangalore.

MAJOR JAMES H. HARROP (*Associate Fellow*), formerly Designer with Avro Aircraft Ltd., Malton, is now Planning Supervisor (Special Projects) with Atomic Energy of Canada Ltd.

W. J. HOBBS (*Associate Fellow*), formerly an Engineer, Orenda Engines Ltd., is now an Assistant Lecturer, Bath Technical College.

J. V. IORNS (*Associate*), formerly with Avro Aircraft Ltd., is now a Senior Aircraft Designer, Lockheed Aircraft Corp., Georgia.

FLT. LT. B. R. JEWELL (*Graduate*), formerly at the Armament Department, R.A.E., is now attending the two year Engineering Course at the College of Aeronautics.

P. G. KALUAGHER (*Student*), formerly with Folland Aircraft Ltd., Hamble, is now an Operational Research Officer, B.O.A.C. at London Airport.

LT. CMDR. R. A. LANGLEY (*Associate Fellow*), formerly Senior Engineer at R.N. Air Station, Culdrose, is now Flag Officer Air (Home), at Lee-on-Solent.

C. R. S. MARSHALL (*Associate Fellow*), formerly Systems Engineer, Technical Design Department, Avro Aircraft Ltd., Ontario, is now with the Advanced Project Group, Hawker Siddeley Aviation Ltd.

JOHN G. MARSHALL (*Associate Fellow*), formerly with Avro Aircraft Ltd., is now on the Engineering staff of the Lockheed Aircraft Corp., Georgia.

R. MASCALL (*Associate*), formerly with Avro Aircraft Ltd., Malton, is now a Liaison Engineer with Canadair Ltd., Montreal.

FLT. LT. R. C. MORING (*Associate*) has been posted from flying duties, R.A.F. Transport Command, Dishforth, to No. 36 Squadron, Transport Command, R.A.F. Colerne, Wilts., on flying duties.

D. E. MORRIS (*Fellow*), formerly Chief Superintendent, A. and A.E.E., is now Chief Superintendent and Head of Aerodynamics Divisions, R.A.E., Bedford.

WING CMDR. R. V. MOXEY (*Associate Fellow*), formerly in the Directorate of Aircraft Engineering, Air Ministry, is now Command Training Officer, Maintenance Command, R.A.F., Andover.

GROUP CAPT. V. C. OTTER (*Associate Fellow*), formerly Command Engineer Officer at Headquarters, Bomber Command, is now Officer Commanding No. 32 Maintenance Unit, Barry, Glam.

J. H. PAUL (*Graduate*), formerly with Avro Aircraft Ltd., is now Assistant Project Engineer, Curtiss-Wright Corp., Caldwell, New Jersey.

A. C. RATTLE (*Associate Fellow*), formerly a Draughtsman, Short Bros. and Harland Ltd., is now Assistant Lecturer in Mechanical Engineering, Preston Technical Institute, Brighton.

CMDR. R. H. REYNOLDS (*Associate*), formerly of H.M.S. *Contest* has been appointed Commander (Air), H.M.S. *Ark Royal*.

T. ROBERTS (*Associate Fellow*), formerly an Aerodynamicist, Avro Aircraft Ltd., Malton, is now an Aeronautical Research Engineer, Space Task Group, N.A.S.A., Langley Field.

R. B. SHANNON (*Associate Fellow*), formerly a Design Engineer, Avro Aircraft (Canada) Ltd., is now a Senior Design Engineer (Armament), North American Aviation, Columbus Division.

J. A. H. SHEPPERD (*Associate Fellow*), formerly Lecturer in Mathematics, Hull University, is now a Lecturer in Mathematics, Manchester College of Science and Technology.

H. W. SIDWELL (*Associate Fellow*), formerly Director and General Manager, Air Service Training Ltd., Hamble, is now Works Director, Gloster Aircraft Co. Ltd., Gloucester.

W. D. SPROSON (*Associate Fellow*), formerly Senior Stress Engineer, Avro Aircraft (Canada) Ltd., Malton, is now Structures Engineer, Convair Aircraft Corp., San Diego.

E. SZOMANSKI (*Associate Fellow*), formerly Staff Research Engineer, Fairchild Engine and Airplane Corporation, New York, is now Staff Engineer (Fluid Dynamics) with the Advanced Technology (Mechanical Engineering) Department of the International Business Machines Company, Product Development Laboratory at Endicott, New York.

P. WESTMORELAND (*Associate Fellow*), formerly a Structures Engineer with Blackburn Aircraft Ltd., Brough, is now a Senior Technical Engineer with the Fixed Wing Group, Bristol Aircraft Ltd., Filton.

LECTURE THEATRE APPEAL

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Previous total (including minor adjustments)	7472	10	1			
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S. A. Andrews, Esq.	10	0	0			
C. Armstrong, Esq.	2	0	0			
Aviation Developments Ltd.	21	0	0			
Capt. W. B. Axford	5	0	0			
E. G. Barber, Esq	2	2	0			
E. L. T. Barton, Esq. ..	2	2	0			
D. F. Beauland, Esq.	1	1	0			
J. S. Bell, Esq.	1	2	0			
S. G. Bendell, Esq.	1	10	0			
A. E. Berriman, Esq. ..	3	3	0			
Prof. W. G. Bickley ..	2	2	0			
F. W. R. Bird, Esq.	2	2	0			
Lt. Col. L. V. S. Blacker ..	1	1	0			
W. B. H. Blydenstein, Esq. ..	3	0	0			
Flt. Lt. H. C. Boyne	12	0	0			
Bristol Branch	25	0	0			
Group Capt. A. F. Britton ..	5	5	0			
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R. S. Brown, Esq	3	3	0			
Bryans Aeroquipment Ltd. ..	5	5	0			
Wg. Cdr. J. J. Carroll ..	2	0	0			
G. R. Clary, Esq	2	2	0			
D. M. Cockburn, Esq. ..	5	0	0			
C. L. Cowdrey, Esq.	10	0	0			
R. C. Cross, Esq.	20	0	0			
H. Davies, Esq.	2	2	0			
Flg. Off. W. F. Dawson	1	1	0			
A. E. Dennard, Esq.	2	0	0			
F. J. W. Wingfield Digby, Esq.	1	1	0			
F. E. Donovan, Esq.	2	2	0			
Wg. Cdr. B. E. Dunkley ..	2	2	0			
H. T. Ellingworth, Esq. ..	1	1	0			
S. A. Embley, Esq.	2	2	0			
A. C. Ericsson, Esq.	1	1	0			
W. P. Escreet, Esq.	2	2	0			
I. A. Fisher, Esq.	1	0	0			
F. A. Foord, Esq	5	0	0			
Sqn. Ldr. L. M. V. Fox-Strangways	3	0	0			
R. A. Fuller, Esq.	1	0	0			
Wg. Cdr. G. L. Gandy ..	5	5	0			
Flt. Lt. N. Gardner	1	1	0			
W. D. Goodwin, Esq. ..	1	1	0			
I. Grant-Murray, Esq. ..	6	6	0			
F. Green, Esq.	10	0	0			
J. C. Green, Esq.	4	0	0			
N. Gregory, Esq.	2	2	0			
P. Grootenhuis, Esq.	3	3	0			
Wg. Cdr. T. E. Guttery ..	2	0	0			
J. E. Gwilliam, Esq.	1	0	0			
R. E. Hardingham, Esq. ..	5	0	0			
R. D. M. Harper, Esq. ..	2	2	0			
M. H. Harries, Esq.	5	0	0			
A. M. Hickling, Esq. ..	1	0	0			
Major J. V. Holman	1	1	0			
Sqn. Ldr. F. A. Houghton ..	2	0	0			
H. B. Howard, Esq.	5	5	0			
O. W. Humphreys, Esq. ..	10	10	0			
M. A. Hundleby, Esq. ..	2	0	0			
Sqn. Ldr. W. E. B. Hurst ..	2	2	0			
Wg. Cdr. E. H. Jenkins ..	2	2	0			
Flt. Lt. B. R. Jewell	1	0	0			
Wg. Cdr. J. Jewell	2	2	0			
A. C. P. Johnstone, Esq. ..	2	2	0			
F. A. King, Esq.	2	2	0			
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J. R. C. Lane, Esq.	2	2	0			
F. S. Lester, Esq.	2	0	0			
D. R. Lewis, Esq.	3	3	0			
Dr. D. E. Lloyd	2	2	0			
P. G. Lucas, Esq.	5	0	0			
B. R. Luesley, Esq.	1	0	0			
M. Madders, Esq.	2	2	0			
Brig. Gen. Sir H. O. Mance	5	5	0			
W. Markowski, Esq.	2	0	0			
S. Matthews, Esq.	1	1	0			
L. W. Meyer, Esq.	2	2	0			
F. G. Miles, Esq.	10	0	0			
R. D. Milne, Esq.	1	1	0			
M. B. Morgan, Esq.	10	10	0			
S. F. Mortimer, Esq.	3	3	0			
Air Chief Marshal Sir T. McEvoy	5	5	0			
J. H. McFarlane, Esq. ..	1	0	0			
National Luchtvaartlaboratorium	2	0	0			
G. H. M. Nayler, Esq. ..	2	2	0			
L. B. Naylor, Esq.	1	0	0			
W. Nowak, Esq.	2	0	0			
A. Ogilvie, Esq.	20	0	0			
Air Vice-Marshal W. A. Opie	2	2	0			
R. K. Page, Esq.	2	2	0			
S. L. Page, Esq.	1	1	0			
Major-Gen. H. T. Parham ..	2	2	0			
R. J. Parkhouse, Esq. ..	2	2	0			
Air Cdre. G. J. C. Paul ..	2	2	0			
S. F. Paul, Esq.	1	1	0			
Prof. E. B. Pearson	1	0	0			
Wg. Cdr. L. C. N. Penney ..	2	2	0			
C. A. Pike, Esq.	15	0	0			
N. L. Prewett, Esq.	1	1	0			
R. W. Potts, Esq.	1	1	0			
Group Capt. J. H. Powle ..	5	5	0			
R. Rathbone, Esq.	1	1	0			
R. Rhodes, Esq.	5	0	0			
S.F.I.M. (Gt. Britain) Ltd. ..	25	0	0			
W. R. Shapey, Esq.	3	3	0			
M. A. Siddigui, Esq.	2	0	0			
Silver City Airways	52	10	0			
Sqn. Ldr. W. M. Simms ..	1	1	0			
G. T. Smith, Esq.	2	0	0			
H. Soloniewicz, Esq. ..	2	0	0			
P. C. G. Stalkartt, Esq. ..	5	0	0			
Flt. Lt. G. W. Staples ..	1	1	0			
V. T. Stevenson, Esq. ..	1	0	0			
H. J. Stieger, Esq.	5	5	0			
A. G. Store, Esq.	2	2	0			
R. B. Stratton, Esq.	5	0	0			
G. J. Sturgess, Esq.	1	1	0			
I. M. Sutherland, Esq. ..	1	1	0			
M. P. Sutton, Esq.	2	2	0			
M. E. Thomas, Esq.	2	2	0			
Flt. Lt. A. B. Thompson (additional)	1	0	0			
E. W. H. Thompson, Esq. ..	2	0	0			
Flt. Lt. R. C. Travis	1	0	0			
Lt. Cdr. W. J. Tuck	2	2	0			
T. G. Turner, Esq.	5	0	0			
C. F. Uwins, Esq.	10	0	0			
Prof. H. J. Van der Maas ..	1	1	0			
Group Capt. J. B. Veal ..	2	2	0			
H. F. Vessey, Esq.	3	0	0			
Dr. W. A. J. Wall	2	0	0			

Donations—*Continued*

	£	s.	d.	£	s.	d.
Dr. B. N. Wallis	1	1	0			
Dr. J. Walton	2	10	0			
A. J. Wangford, Esq.	2	2	0			
Wg. Cdr. R. A. Watts ..	5	5	0			
D. Webb, Esq.	2	2	0			
Weybridge Branch	14	7	0			
B. C. Whorwood, Esq. ..	5	0	0			
B. J. Williams, Esq.	2	2	0			
J. C. Wimpenny, Esq. ..	5	5	0			
H. M. Woodhams, Esq. ..	5	5	0			
M. Wotton Wood, Esq. ..	50	0	0			
S. D. Wynn, Esq.	5	0	0			
				£8109	12	1

7-year Covenants

	£	s.	d.	£	s.	d.
Previous total (including minor adjustments)	1970	p.a. 2	9			
T. E. B. Bateman, Esq. ..	1	0	0			
Dr. J. Black	1	1	0			
L. J. S. Bradbury, Esq. ..	1	10	0			
L. J. Bristow, Esq.	1	1	0			
C. H. Burge, Esq.	1	0	0			
S. M. Cobb, Esq.	1	0	0			
A. F. Constantine, Esq. ..	1	1	0			
Wg. Cdr. J. Crane	1	1	0			
K. G. Dunn, Esq.	1	0	0			
Sir George R. Edwards ..	10	10	0			
Sqn. Ldr. J. C. Ellis	1	1	0			
V. W. Erlebach, Esq. ..	1	0	0			
Sir William Farren	5	0	0			
J. R. C. Fearon, Esq. ..	1	1	0			
J. W. Fozard, Esq.	1	1	0			
H. H. Gardner, Esq.	5	0	0			
M. R. Gordon-Clark, Esq. ..	5	0	0			
J. D. L. Gregory, Esq.		10	6			
Lt. Cdr. R. P. Grimwood ..	1	0	0			
G. L. Gunstone, Esq. ..	1	1	0			
Flt. Lt. J. L. Gutteridge ..	1	0	0			
W. W. Hackett, Esq.	5	0	0			
E. Handley, Esq.	2	0	0			
J. H. Hawthorne, Esq. ..	1	0	0			
J. H. Heap, Esq.	1	0	0			
Plt. Off. D. M. K. Houston ..	1	0	0			
M. S. Igglesden, Esq. ..	1	1	0			
L. W. E. Johnston, Esq. ..	2	0	0			
H. E. Jones, Esq.	5	0	0			
D. E. Lennard, Esq.	3	0	0			
M. Lewin, Esq.	2	2	0			
J. D. Lewis, Esq.	1	1	0			
G. M. Lilley, Esq.	1	1	0			
Joseph Lucas (Industries) Ltd.	437	10	2			
N. W. McCue, Esq.	1	1	0			
Wg. Cdr. I. N. M. McDonald	2	0	0			
R. Maine, Esq.	1	0	0			
Flg. Off. O. Maung	2	0	0			
R. Melling, Esq.	3	0	0			
E. Miller, Esq.	2	0	0			
R. J. Monaghan, Esq. ..	2	0	0			
D. W. Morgan, Esq.	1	0	0			
F. W. Morgan, Esq.	1	0	0			
Air Cdre. J. R. Morgan ..	1	0	0			
P. C. F. Morgan, Esq. ..	1	0	0			
E. N. Morton, Esq.	1	10	0			
Air Cdre. H. E. Nowell ..	2	2	0			
P. C. Parks, Esq.	2	0	0			
T. J. Patrick, Esq.	1	0	0			
G. H. Pennington, Esq. ..	1	0	0			
Flt. Lt. J. D. Penrose ..	1	1	0			
R. H. Platt, Esq.	1	0	0			
D. Rendel, Esq.	1	0	0			
J. M. Roberts, Esq.	1	1	0			
J. Schofield, Esq.	2	0	0			
E. Shipley, Esq.	1	0	0			
J. H. Sinclair, Esq.	1	0	0			
H. C. Smith, Esq.	2	2	0			
R. L. E. Toms, Esq.	2	0	0			
D. E. Turner, Esq.	1	0	0			
Rear-Admiral A. J. Tyndale-Biscoe	1	0	0			
L. S. Wigdor, Esq. (additional)	3	19	0			
J. W. Wilson, Esq.	2	2	0			
P. W. Wreford, Esq.	1	0	0			
				*£2517	15	5

10-year Covenant

Previous total				*£612	10	0

**With Income Tax at the present standard rate of 7s. 9d. in the £, these figures should produce £38,750 (approx.) over the 7- and 10-year periods.*

Bankers' Orders for an Unspecified Number of Payments

	£	s.	d.	£	s.	d.
Previous total	47	1	0			
A. W. Cardrick, Esq. ..		15	0			
Mrs. M. A. Hyde	1	0	0			
R. A. Kirkin, Esq.	1	1	0			
Sqn. Ldr. A. C. Sanderson ..	5	0	0			
I. H. Smart, Esq.	5	5	0			
				£60	2	0
				£11,299	19	6

GRAND TOTAL (Approximately) **£47,000**

(allowing for recovery of income tax on 7 and 10 year covenants)

Promises amounting to another £25,000 had been received at the time of going to press.

The following have also subscribed but do not wish the amounts of their contributions to be published

F. W. Armstrong, Esq.
C. Baxter, Esq.
Miss B. M. Bigg
C. T. Bradshaw, Esq.
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J. Yeeles, Esq.

The Journal of the Royal Aeronautical Society

WITH WHICH IS INCORPORATED THE INSTITUTION OF AERONAUTICAL ENGINEERS

VOLUME 63 SEPTEMBER 1959 NUMBER 585

HYPERSONIC FLOW

An All-Day Discussion on Hypersonic Flow was held by the Society on 15th December 1958, at the Institution of Mechanical Engineers, Birdcage Walk, London S.W.1. Dr. G. W. H. Gardner, C.B., C.B.E., D.Sc., F.R.Ae.S. (now Sir George), Director of the Royal Aircraft Establishment, Farnborough, presided throughout the day. Eight papers were given by:—
Mr. R. J. Monaghan, M.A., F.R.Ae.S., Dr. D. W. Holder, F.R.Ae.S., Dr. R. N. Cox, M.A., B.Sc., Dr. K. W. Mangler, F.R.Ae.S., Dr. L. F. Crabtree, B.Sc., M.I.A.S., A.F.R.Ae.S., Professor M. J. Lighthill, F.R.S., Dr. J. A. Shercliff, M.A., S.M. and Mr. T. R. F. Nonweiler, B.Sc., A.F.I.A.S., A.F.R.Ae.S. Miss Beryl E. Beadle, B.Sc.(Eng.), M.I.A.S., A.F.R.Ae.S., acted as rapporteur for the Discussion.

Problems of Hypersonic Aerodynamics—A Survey

by

R. J. MONAGHAN, M.A., F.R.Ae.S.
(Aerodynamics Department, Royal Aircraft Establishment)

SUMMARY: Real gas effects are among the major complications of hypersonic flow and this is illustrated by examples of the temperatures and pressures attained when a flow is brought to rest. Once dissociation appears, these depend to a marked extent on the type of compression.

In designing for the alleviation of aerodynamic heating there are two distinct cases. The first is the uncontrolled re-entry of a freely falling body, for which it is best to have a shape with a high pressure drag. The second is sustained flight, for which a shape with low pressure drag could be better, radiation giving appreciable control of surface temperature.

Low pressure drag accords with design for aerodynamic efficiency and there is scope for research on three-dimensional lifting shapes. There is some discussion of this and also of the philosophy of securing maximum amounts of separated flow.

Finally, there is continual emphasis on the need for experimental research, since hypersonic flow fields may differ considerably from those that would be expected by extrapolation of conventional supersonic experience.

1. Introduction

The term "hypersonic" is commonly used to describe flight at Mach numbers greater than about 5, this being a region where the linearised equations of supersonic flow are no longer adequate, where increased boundary layer thicknesses interact appreciably with flow fields around slender bodies and where aerodynamic heating assumes major importance in missile or aircraft design.

A further boundary occurs at Mach numbers of the order of 8 to 9, when there is sufficient kinetic energy to provide some dissociation of the oxygen molecules in air and, as speeds are increased above this value, appreciable proportions of the constituents of air may be dissociated (and ionised) by passage through the normal shock wave ahead of the stagnation point of a body. This stagnation point will always exist, even on bodies or wings with nominally sharp tips or leading edges.

Even at points well away from the tip or leading edge, where the local inviscid flow may not be dissociated, the boundary layer of retarded air provides the means of dissociating the air close to the surface of the body.

Once dissociation has appeared, one is dealing no longer with the "ideal gas" of classical aerodynamics, which assumes that the ratio of the specific heats (γ) is constant. Instead, we are dealing with a real gas composed of molecules and atoms (and ions), whose concentrations may vary from one point to another and new species, such as NO, can make their appearance.

The paper by Professor Lighthill considers the physics of these real gases. Meanwhile, a convenient introduction to their impact on the aerodynamics of a situation is given by considering the variation of stagnation temperature with flight speed, assuming that the gas is always in chemical equilibrium.

2. "Stagnation" Temperature Rise

Figure 1 shows "stagnation" temperature in °K plotted against flight speed in ft./sec., at an altitude of 200,000 ft. in the atmosphere.

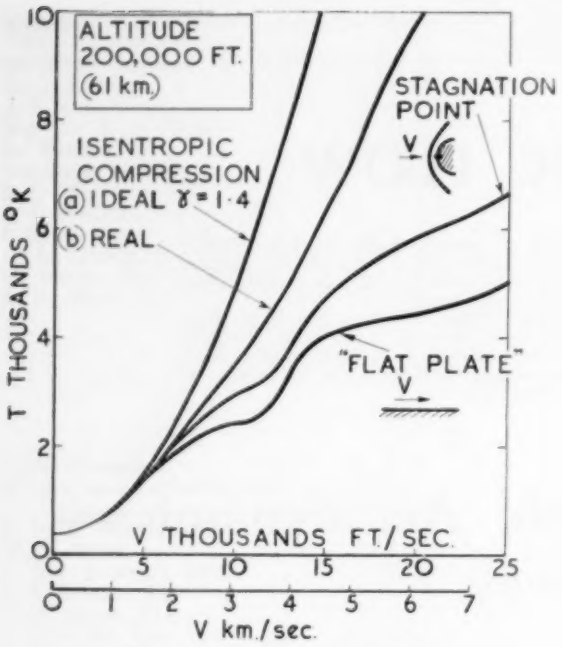

FIGURE 1. Stagnation temperatures.

At low speeds we are accustomed to the useful formula that the stagnation temperature rise in °C is given by $(V/100)^2$, where V is the flight speed in miles per hour. This results in the topmost curve in Fig. 1, which would give a stagnation temperature exceeding 10,000 °K at a flight speed of 15,000 ft./sec.

Now in fact, it is the stagnation enthalpy rise (not the temperature rise) that varies with the square of the flight speed and the next curve shows how this affects the stagnation temperature of air, treated as a real gas. At speeds above 3,000 ft./sec. the real gas curve drops away noticeably from the ideal gas curve, because of the increase in specific heat arising from excitation of the vibrational mode in the molecules.

At speeds above 6,000 ft./sec., dissociation introduces a further complication. It absorbs energy and hence reduces the temperature rise. The amount of energy absorbed depends on the degree of dissociation and, considering conditions of chemical equilibrium, the degree of dissociation depends on density as well as on temperature.

Hence, it becomes necessary to define the type of compression and several cases are considered in Fig. 1. Isentropic compression is the most efficient from the point of view of pressure recovery and the corresponding curve shows that the isentropic stagnation temperature at 15,000 ft./sec. is about 6,500°K rising to 10,000°K at 20,000 ft./sec.

It is relevant to consider the corresponding stagnation pressures since these are the pressures that could be required in the settling chamber of a conventional wind tunnel. The ambient pressure at 200,000 ft. was taken to be $3{\cdot}7 \times 10^{-4}$ atmospheres. Then the ideal gas stagnation pressure for a flight velocity of 15,000 ft./sec. would be 60 atmospheres, whereas in the real gas case it would be about 500 atmospheres. This marked increase, brought about by real gas effects, is an indication of the difficulties encountered by the wind tunnel designer if he seeks full simulation of hypersonic flight conditions. It is also an indication of the large scale of the changes introduced by real gas effects.

The next curve shows the temperature that could be obtained at the stagnation point of an insulated body (neglecting radiation). In this case the air passes through a normal shock wave and is then compressed isentropically to the stagnation point. Because of the shock wave, the pressure recovery is lower than with isentropic compression: lower pressure means more dissociation and hence lower temperature for a given enthalpy. For example, at 15,000 ft./sec., the temperature is now about 4,750°K corresponding to a pressure of 0·075 atmospheres. The reduction in pressure is striking.

Finally the lowest curve shows the temperatures corresponding to the ambient pressure of $3{\cdot}7 \times 10^{-4}$ atmospheres. These might correspond to the zero heat transfer temperatures on a flat plate, with a recovery factor of unity, and assuming that the plate has an ideally sharp leading edge. In this case the temperature at 15,000 ft./sec. is just on 4,000°K.

Thus we see how stagnation temperature is modified by pressure when dissociation is occurring. Therefore, changing the altitude will affect these stagnation temperatures: at 100,000 ft. the stagnation temperatures at the upper end of the speed range would be increased by about 1,000°C. Also the isentropic stagnation pressure corresponding to a flight velocity of 15,000 ft./sec. would be in excess of 10,000 atmospheres.

A paper by Bird[1] gives further examples of the influence of real gas effects.

Finally, there is an interesting feature that is shown up well by the flat plate curve. The first "knee" on this curve, at around 10,000 ft./sec. is the result of oxygen dissociation; the next "knee", around 15,000 to 20,000 ft./sec. is caused by nitrogen dissociation, and we may note that the oxygen dissociation is essentially complete before nitrogen dissociation begins. This leads to some simplification in analytical work. Also over the speed range considered, ionisation has negligible effect on the energy balance.

3. Design for the Alleviation of Aerodynamic Heating

Considering design for flight at hypersonic speeds, it is apparent that these stagnation or zero heat transfer temperatures are too high for comfort and we must seek the means of minimising the temperature that the surface of a body would reach. This involves knowledge of aerodynamic heating rates and there will be some discussion of these in the paper by Dr. Crabtree. However, before leaving Fig. 1 it must be emphasised that the temperatures on it are not representative of the "driving temperatures" behind heat transfer in the individual cases considered. When there is dissociation, energy is transferred by the diffusion of atoms across

streamlines (carrying with them their dissociation energies) as well as by normal heat conduction. As a result, heat transfer is related to total enthalpy rather than to total temperature, and, for a given flight speed, the total enthalpy is the same for all the cases considered in Fig. 1.

In considering design for the alleviation of aerodynamic heating, two distinct cases emerge. The first is the uncontrolled entry or re-entry into the earth's atmosphere of a freely falling body, for example, after a ballistic trajectory. The second is that of sustained flight within the atmosphere, which includes hypersonic boost-glide vehicles or controlled re-entry from space.

The problem of the freely-falling body has been treated in some detail by Allen and Eggers[2] and by other authors and I do not propose to say much about it here. The important conclusion reached by Allen and Eggers is that if the body is relatively light and is slowed down considerably in traversing the atmosphere, then a blunt shape with high pressure drag can offer the best promise of minimising the overall heat input. The philosophy in this case is that on re-entry, the body possesses a large amount of kinetic energy and the difference between this and the residual kinetic energy at impact will be dissipated as heat as the body slows down in traversing the atmosphere. Now aerodynamic heating is related to the work done against skin friction, while work done against pressure drag produces heat only in the wake: hence it is desirable to keep the ratio of skin friction to pressure drag small, as is the case with the bluff body.

If, on the other hand, we consider the case of sustained flight, then high drag shapes would not seem to tally with the desire for a reasonable lift-drag ratio and a fresh approach is required.

Since, by definition, the flight is likely to last some time, it seems logical to enquire what relief we could expect from radiation. Thus, if the lift coefficient is large enough to permit flight at very high altitudes, then the reduced aerodynamic heating rates might be balanced by radiation before the surface temperature became unduly high.

This is examined in Fig. 2, which shows flight conditions that could give an equilibrium temperature of 1,500°K on various bodies, assuming a surface emissivity factor of 0·9. The plot is of Reynolds number per foot, based on ambient conditions and flight speed, against flight speed in ft./sec. Curves of constant altitude (ft.) are plotted across the graph.

The various bodies are a hemisphere (representative of the high drag shape), and a sharp-nosed cone and an ideal flat plate (representative of low drag shapes). The temperature is a mean temperature related to a reference length of one foot and the boundary layers were taken to be laminar or turbulent in accordance with the Reynolds numbers involved.

The interesting feature is that the low drag shape turns out to be better than the high drag shape in that higher speeds are possible for a given altitude. For example, at 120,000 ft. the hemisphere is limited to 10,000 ft./sec., but the ideal flat plate could exceed 17,000 ft./sec. A second feature is that Reynolds

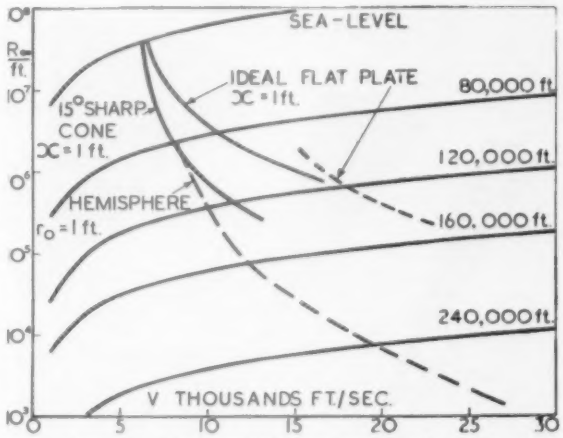

FIGURE 2. Mean equilibrium temperature 1,500°K.

numbers and speeds are such that the flows would mostly be in the continuum region. However, if the permitted temperature were reduced from 1,500°K the operating altitudes would have to be increased.

I should add that in a recent paper[3], Nicholson reaches somewhat similar conclusions when considering the re-entry problem. Thus, if the time of re-entry can be stretched from seconds to minutes, by the use of aerodynamic lift, then radiation from a surface at tolerable temperature could be sufficient to balance the aerodynamic heat input. A later paper, by Mr. Nonweiler, will be considering further aspects of the re-entry problem.

4. Design for Aerodynamic Efficiency

Turning to the consideration of design for aerodynamic efficiency and whether this reconciles with requirements for alleviating aerodynamic heating, we derive some hope from the results shown in Fig. 2. However, these are mean temperatures over a length of the body and there is the immediate question of how to protect the leading edge, or tip, where the heating rates can be very high. Also, it has been assumed that the plate has a sharp leading edge, with attached plane shock wave and no pressure gradients along its length. Similar assumptions are made for the cone. As Dr. Cox will show in his paper, experiments show that this happy state of affairs just does not exist in the speed range of interest in Fig. 2. On flat plates with nominally sharp leading edges, the leading edge shock is detached and curved, and there are considerable variations in pressure along the length of the plate. The chordwise extent of this pressure variation is related to leading edge thickness but since the leading edge effect upsets the aerodynamics in any case, it seems relevant to enquire what penalties would be occurred by designing deliberately for blunt leading edges, which would be more suitable for dealing with the aerodynamic heating problem. The "ideal flat plate" curves in Fig. 2 would then move to the left, but would still be better than the hemisphere.

This leading edge effect is an instance of a novel

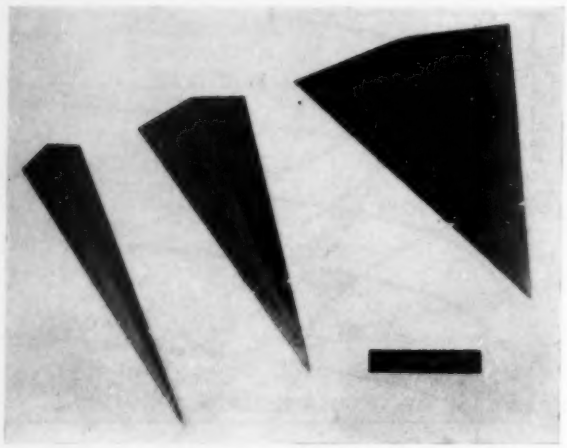

FIGURE 3. Arrowhead models for the R.A.E. Hypersonic Tunnel.

feature being shown up by experimental research on hypersonic flow. It emphasises the need for experimental research and indicates that errors could arise if one based one's hypersonic ideas solely on extrapolation of conventional supersonic experience.

Sweepback would help in reducing both the drag and the leading edge heating and it is interesting to speculate on the behaviour of a genuinely three-dimensional lifting shape at hypersonic speeds. In general the Mach number would be too high for this shape to satisfy the requirements of an aerodynamically slender body in the sense used at low supersonic speeds, but it is interesting to speculate whether any of the slender body characteristics of supersonic flow could be realised at hypersonic speeds. For example, would it be possible to shed vorticity from the leading edge of a highly swept delta shape at incidence? If so, then some of the heating problem at the leading edge might disappear. For reasons such as these, we are making some tests in the R.A.E. Hypersonic Tunnel, at $M = 7$, on arrowhead shapes with sweepback angles of 70°, 80° and 85° (Fig. 3), *i.e.* the range of angles covers supersonic through to subsonic leading edges. At higher Mach numbers it becomes practically impossible to have a leading edge subsonic relative to the Mach lines of the undisturbed flow, but there can still be argument as to how a supersonic leading edge at high incidence would behave.

For the first tests, the arrowheads have a symmetrical spanwise section. There could obviously be an advantage in introducing camber and, from here on, discussion can range widely and freely on other modifications in shape that might be advantageous at hypersonic speeds.

5. Separated Flows and Magnetogasdynamics

If one tackles the design problem from the alternative viewpoint of minimising aerodynamic heating and then seeking aerodynamic efficiency, there is much to be said for the philosophy of securing maximum amounts of separated flow. An instance here is the well known flow separation provoked by a spike in front of a bluff body which reduces its drag. By contrast, the overall heat transfer may be increased. This can happen if transition to turbulence takes place along the boundary of the separated flow bubble, whereas on the original body without spike the boundary layer stayed wholly laminar. Also, recent research at N.P.L. and R.A.E. shows that high local heating rates can be experienced in regions of re-attachment (as will be discussed by Dr. Crabtree). So, the separated flow philosophy is not as simple as might be thought at first sight, and there is interest in both the bubble and the vortex types of flow separation.

Before leaving this topic, one should mention the fact that the ionisation of the air produced by flight at very high speeds raises the possibility that it might be possible to exert some control, at least on the boundary layer, by electromagnetic forces. This is a new field and Dr. Shercliff will be discussing it in his paper.

6. Conclusion

The scope of the present survey paper has been restricted to some of the fluid motion problems that beset the aerodynamicist when he considers the possibilities for flight at hypersonic speeds. I am well aware that I have omitted many major problems; for example there has been no mention of problems of ablation and sweat-cooling, or of the problems introduced by possible chemical reactions either at the surface or in the boundary layer. For discussion of problems of performance and propulsion, I would refer you, for example, to a recent paper by J. E. Allen[4]. (The latter contains 98 references). What I have tried to do is to show:—

(*i*) that real gas effects (notably dissociation) can have considerable effect on hypersonic flow fields, and no longer can we take for granted the properties of our working fluid,

(*ii*) that aerodynamic efficiency and protection against aerodynamic heating are not necessarily incompatible with each other, and

(*iii*) that theory must be backed by experiment, since hypersonic flow fields may differ considerably from those expected from extrapolation of supersonic experience.

I would stress the last point on the need for controlled experiments and the paper by Dr. Holder considers the new facilities and new measuring techniques with which we are becoming involved.

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Experimental Facilities and Measuring Techniques

by

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SUMMARY: Some of the aerodynamic problems that arise in flight at hypersonic speeds can be studied experimentally with wind tunnels and measuring techniques that do not differ in principle from those used for research on supersonic flow. If, however, it is required to simulate the very high temperatures of hypersonic flight, it is usually necessary to use heaters and tunnels of unconventional design, frequently having a very short running time, or to make tests with a model launched or propelled at high velocity.

The short running times sometimes involve the use of measuring techniques that differ considerably from those of conventional wind tunnel practice. Also, in the presence of high temperature, real gas effects, it is sometimes necessary to measure additional quantities in order to define the state of the gas. The paper contains a brief introductory review of these topics, and of the extent to which experimental test facilities can reproduce the conditions of full-scale hypersonic flight.

1. Introduction

As Mr. Monaghan has indicated in his introductory paper, flight at hypersonic speeds involves several new aerodynamic problems. This paper gives a brief general description of some of the methods that may be used to study these problems experimentally, with the object of providing a framework for detailed points that may be raised during the discussion.

NOTATION

ρ	density
V	velocity
T	absolute temperature
h	specific enthalpy
$\mathscr{R}$	gas constant
R	Reynolds number
M	Mach number
l	typical linear dimension

Suffixes

0	atmospheric conditions at sea level
H	stagnation conditions.

2. Important Aerodynamic Parameters and their Full-Scale Values

When designing an experiment it is necessary to estimate the parameters that are likely to be important, and the range of these parameters that should be covered to provide results of full-scale interest. For a given test gas, the important aerodynamic parameters are likely to be the Mach and Reynolds numbers, the specific enthalpy, and the density, although in some tests simulation of all of these quantities may not be necessary. For example, in measurements near the stagnation points of bluff bodies, Reynolds number and Mach number may be relatively unimportant. On the other hand, if the rates of dissociation and ionisation are not so rapid that the flow is everywhere effectively in thermal equilibrium, additional parameters* need to be considered, and it may then prove impossible to simulate the full-scale conditions completely on a reduced-scale model. There is some evidence that such non-equilibrium effects are often relatively unimportant, but more work is required before a definite statement on this point, or on the importance of the other parameters for different types of body configuration and flight conditions, can be made.

As examples of the range of the similarity parameters encountered under full-scale conditions, two types of hypersonic vehicle will be considered, assuming that the altitude-velocity relationships and the properties of the atmosphere are known, and that equilibrium conditions exist. Fig. 1 shows the values of the parameters for a hypothetical ballistic missile with a range of 6,000 miles, which gives a re-entry velocity of about 23,000 ft./sec. The velocity falls with decreasing altitude, aerodynamic effects becoming of major importance below about 100,000 ft. There is a corresponding fall of Mach number. The Reynolds number rises with decreasing altitude to a very high value, and then falls again; the trend of variation of stagnation density is similar. The stagnation temperature is very high, being initially about 7,000°K, and so is the stagnation specific enthalpy. Temperature is included as well as enthalpy because the former may be more familiar.

If a hypothetical boost-glide vehicle† is considered the results shown in Fig. 2 are obtained; to aid comparisons, this is plotted to the same scales as Fig. 1. The velocity and Mach number are again high, and so are the stagnation temperature and specific enthalpy. The Reynolds number and density are, however, very low compared with the peak values achieved by the ballistic missile because of the greater altitudes. Static rather than stagnation density is considered here

*For dissociation, the appropriate similarity parameter is of the form $\frac{\rho l}{V}$, and for recombination (assuming a three-body process with the body surface as third body) $\frac{\rho^2 l}{V}$. Here ρ is the density and V the velocity of the flow, and l is a typical body dimension.

†The values shown have been computed for a lift/drag ratio of about 5, and a wing loading of about 20 lb./ft.².

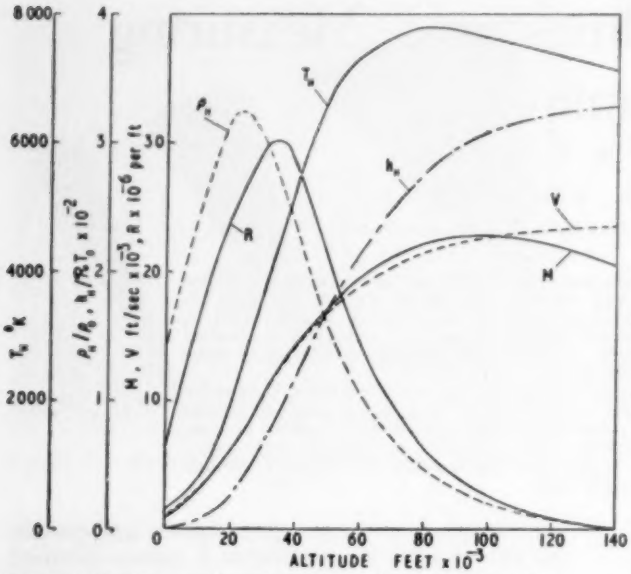


FIGURE 1. Typical aerodynamic conditions for a ballistic missile.

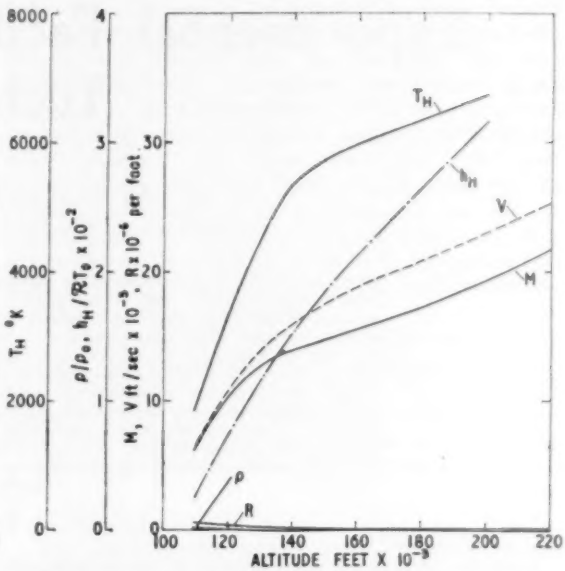


FIGURE 2. Typical aerodynamic conditions for a glide vehicle.

because it is probably more appropriate to the relatively slender configuration that would be used for a glide vehicle.

3. The Main Types of Experimental Facility

Before considering the extent to which these full-scale values may be simulated in experiments, a brief description will be given of the main types of experimental facility that may be used.

3.1. HYPERSONIC WIND TUNNELS AND OTHER FACILITIES WITH LONG RUNNING TIMES

Because the length of this paper is restricted, little description will be given of wind tunnels in which the air stream is heated in order to avoid liquefaction at the working section. Such tunnels[1] are essentially devices for testing under perfect gas conditions, because the temperatures are not high enough to reproduce high temperature, real gas effects such as dissociation. For example, Fig. 3 shows the stagnation temperatures required to prevent liquefaction of air at the working section for a range of stagnation pressures and Mach numbers. It is seen that, for a Mach number of ten, a typical value of the stagnation temperature is about 1,000°K, and for engineering reasons higher temperatures are seldom used in tunnels of this type which may operate continuously, or intermittently for periods of the order of a minute.

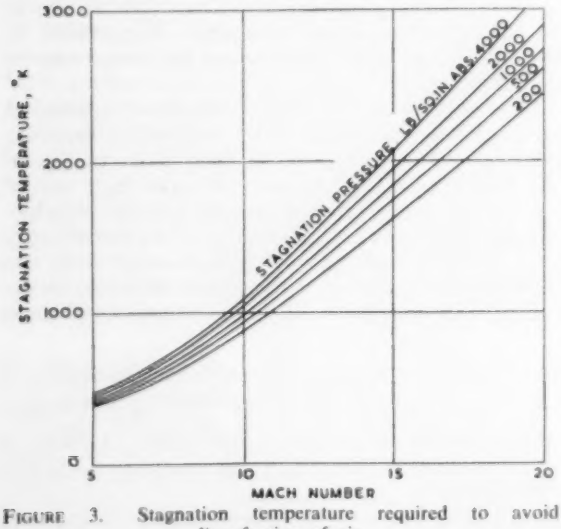


FIGURE 3. Stagnation temperature required to avoid liquefaction of air.

If the simulation of high temperature, real gas effects is not required, a useful alternative to heating air to avoid liquefaction is to use an unheated working gas which does not liquefy until lower temperatures are reached. The most suitable gas for this purpose is helium[2, 3], which may be expanded to Mach numbers in excess of 20 without preheating. Care must be taken to avoid impurities, since the heat release resulting from their condensation will have large effects on the flow in the working section because the enthalpy there is very low. Several wind tunnels using helium have been constructed, and have provided valuable results; uncertainties in the application of these results to the flow of air arise, because helium is a monatomic gas, and further research is required to study these.

Higher temperatures up to, say, 2,500°K have been achieved in wind tunnels or free jets by using special types of storage heater, or by using combustion heating[4]. Such facilities may become expensive if their scale is large, but they provide the long running time at high enthalpy that is probably essential for certain types of structural test.

Even this temperature is, however, too low to

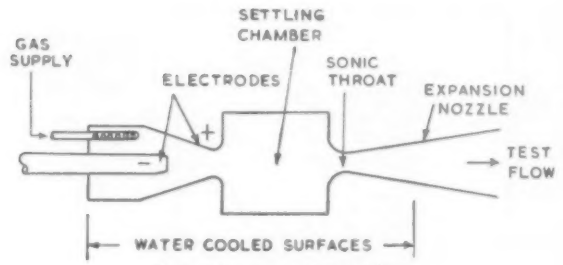


FIGURE 4. An arc-heated jet.

reproduce the high temperature, real gas effects encountered in flight at Mach numbers greater than about ten. The only types of experimental facility with a long running time that have yet been developed to reproduce these temperatures are the arc-heated or so-called "plasma" jets[5,6,7,8]. The principle is illustrated by the typical arrangement sketched in Fig. 4. A d.c. arc is struck between two electrodes, the cathode being a central rod, and the anode having an orifice in its centre. The test gas is supplied to the arc chamber, and, after heating by the arc, escapes through the orifice in the anode. In most arrangements, the flow then enters a settling chamber before expanding in the nozzle used to provide the test stream. A major problem in such facilities, especially when they operate at large power, and with air rather than an inert gas as working fluid, is to reduce electrode (especially anode) erosion and the associated contamination of the test stream. The electrodes are usually made from carbon, tungsten or water-cooled copper, and erosion is usually further reduced by rotating the arc path with respect to the electrodes by introducing the test gas into the arc chamber with swirl, or by means of an external magnetic field.

The electrical power requirements vary with the size of the facility and the gas density from perhaps 50 kilowatts, for a unit used to supply a small low-density wind tunnel, to 10 megawatts, or more, for a unit used to simulate re-entry problems. Uncertainties remain to be resolved concerning the uniformity, thermal equilibrium, and composition of the test stream, and for this reason arc-heated jets have in the past tended to be used for overall studies of ablation and allied structural problems, rather than for detailed aerodynamic research. Many of these uncertainties will, however, no doubt be resolved as a result of further research, and in the author's opinion this type of facility has many attractions, especially, perhaps, for laboratories where heavy electrical power supplies already exist for other purposes.

3.2. FACILITIES WITH SHORT RUNNING TIMES

Other types of facility for the study of high temperature hypersonic flows overcome the engineering difficulties by using operating times that are very short compared with those of conventional wind tunnels. A first step in this direction is to heat the air by a nearly adiabatic compression before it enters the wind tunnel, as in the installation developed[9] at the Polytechnic Institute of Brooklyn.

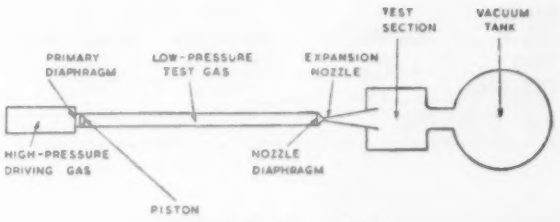


FIGURE 5. A gun tunnel.

3.2.1. *Gun Tunnels*

A similar principle is used in the gun tunnel[10, 11] which is sketched in Fig. 5. Here a light free piston is driven at supersonic velocity along a tube (frequently a gun barrel) by the sudden application of high pressure driving gas to its base, when the primary diaphragm is ruptured. A shock wave is formed ahead of the piston, and reflects from the initially-closed end of the tube back towards the piston. This reflection process is repeated between the piston face and the closed end, and the test gas is heated by shock compression during the process. Eventually, the rise of pressure of the test gas brings the piston to rest, and the space between the piston and the nozzle diaphragm is then filled with gas at high, reasonably uniform, pressure and temperature. The nozzle diaphragm is then allowed to burst, and the hot gas expands in the nozzle and produces a high Mach number stream at the working section. As the flow expands into the nozzle, the piston moves towards the nozzle entry and, in simple theory, the run would end when all of the hot flow had passed into the nozzle, and the piston had reached the nozzle entry. In practice the permissible flow duration is usually limited by melting of the nozzle throat, or by a reduction of stagnation temperature resulting from heat losses from the compressed gas. Flows with real gas stagnation temperatures of about 3,000°K may, however, be achieved for times of the order of 1/10 second, and longer running times are possible at lower temperatures.

3.2.2. *Shock Tubes*

Shock heating is also used to produce the high temperature test gas in the shock tube and the shock tunnel. A shock tube[12, 13] is shown in Fig. 6; it consists of a tube divided initially into two compartments by a diaphragm. The part of the tube to the right of the diaphragm is filled with the test gas at low pressure, and the part to the left of the diaphragm by the driving gas at high pressure. When the diaphragm bursts, the expansion of the driving gas causes a shock wave to propagate along the tube, and as the test gas passes through the shock it is compressed, heated and set in motion in the shock direction. If the shock velocity is constant along the tube, the shock is thus followed by a region of uniform high temperature flow, which may be used for experimental purposes if a model is held in the tube. This uniform flow is terminated by the so-called contact surface, behind which there is the driving gas at relatively low temperature. If the effects of viscosity are ignored, the duration of the uniform flow between the shock and the contact surface increases

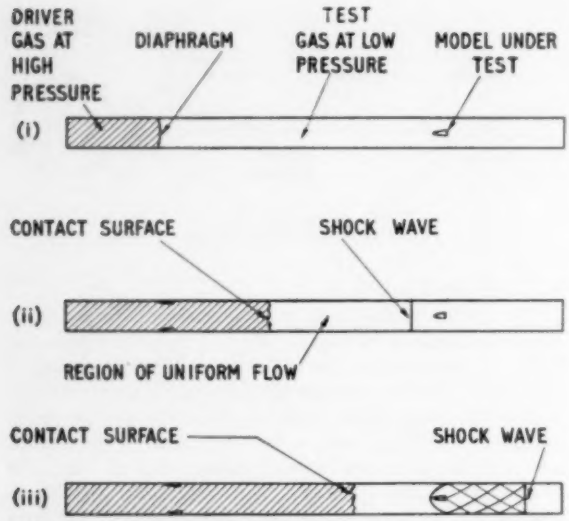


FIGURE 6. Sketches illustrating the operation of a shock tube. (*i*) Initial conditions. (*ii*) After diaphragm bursts. (*iii*) After establishment of flow in the working section.

continuously with increase of the length of the tube because, as indicated in Fig. 7, the shock velocity exceeds that of the contact surface. Because of viscous effects, however, the velocity of the contact surface is increased, and a stage is frequently reached where further increase of the tube length produces little increase in the flow duration. The duration of uniform flow is, in fact, frequently less than a millisecond, but in this time very high enthalpies and densities may be achieved. The Mach number of the air flow behind the shock is, however, limited to about 3 (allowing for real gas effects).

To achieve high shock velocities (or high piston velocities in the gun tunnel), and hence high flow temperature, a light driving gas like helium or hydrogen is used. The shock velocity is sometimes further increased by heating the driving gas either electrically or by combustion. An alternative scheme for obtaining faster shocks for a given overall pressure ratio, at the expense of some loss of running time, is to use the double-diaphragm technique[14]. Here the shock is first propagated into an intermediate gas, such as argon, and then into the test gas when a second diaphragm ruptures.

3.2.3. *Shock Tunnels*

To overcome the Mach number limitation of the shock tube, the shock tunnel was developed. Here, the high temperature flow behind the shock in a shock tube is expanded in a nozzle to high Mach number. Two methods of operation are possible, and these are illustrated in Figs. 8 and 9. In the "straight-through" method[15], the flow behind the shock is expanded directly in a nozzle. The nozzle entry is usually smaller than the diameter of the shock tube, partly to enable the large nozzle area ratio required for the production of high Mach number to be achieved without too large a working section, and partly to permit the boundary

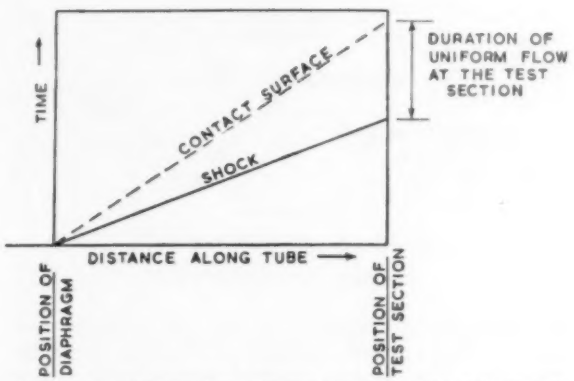


FIGURE 7. Distance-time diagram for the shock and contact surface in a shock tube.

layer on the wall of the driving tube to be by-passed. A certain time elapses after the primary shock strikes the nozzle throat before the system of starting shocks in the nozzle has passed downstream of the model, and left uniform flow past it. This nozzle starting time may be reduced by placing a weak diaphragm ahead of the nozzle lip and evacuating the nozzle and working section to a low pressure. The starting time nevertheless remains an appreciable fraction of the duration of the uniform hot flow provided by the driving tube. Indeed, under certain conditions these two times may become equal, so that no uniform flow is obtained in the expanded working section.

It is seen, therefore, that the duration of the uniform flow at the test section is substantially reduced when an expansion nozzle is fitted, and, since little further reduction* of nozzle starting time is possible, it is of interest to consider methods for increasing the duration of the hot gas supply provided by the driving tube.

The second method[16, 17, 18] of operation illustrated in Fig. 9 enables this to be done. Here the primary shock in the driving tube is reflected from the nozzle inlet (a weak nozzle diaphragm is again included to shorten the duration of the nozzle starting process), and

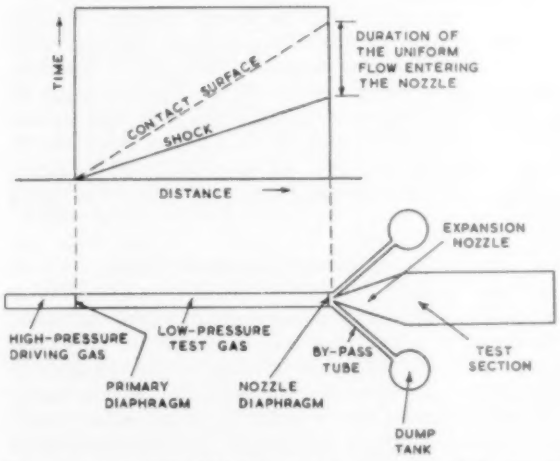


FIGURE 8. A "Straight-through" hypersonic shock tunnel.

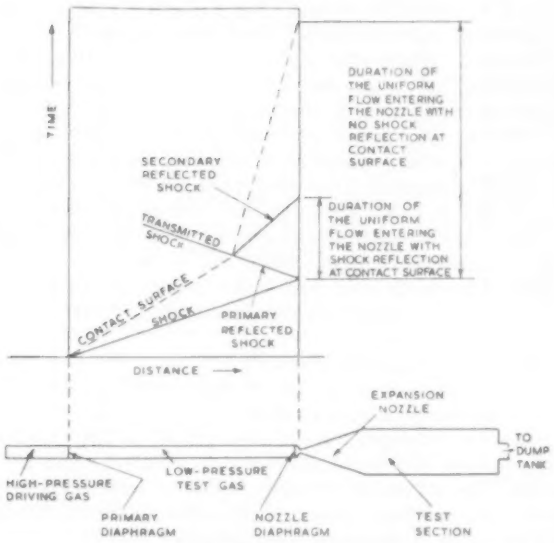


FIGURE 9. A "Reflected-Shock" hypersonic shock tunnel.

leaves a region of almost stationary hot gas between the nozzle inlet and the reflected shock. When the reflected shock meets the contact surface there will, in general, be a secondary reflected shock which moves back towards the nozzle inlet (*see* Fig. 9) and a transmitted shock. The velocity of the contact surface will be greatly reduced on interaction with the primary reflected shock, so that the flow duration, as limited by the arrival of the contact surface in the nozzle, will be substantially increased. The majority of this increased duration is, however, useful only if the disturbance created by the secondary reflected shock is too weak to have a significant effect on the uniformity of the flow entering the nozzle. In theory, the secondary reflected shock may be eliminated by using[19] a so-called "tailored" contact surface. Here the initial conditions in the tube are arranged so that the primary reflected shock is wholly transmitted on striking the contact surface. For given driving and test gases, and given initial temperatures, this condition can be achieved strictly at only one value of the primary shock Mach number (*i.e.* only one value of the stagnation specific enthalpy of the test stream), so that a range of driving gases or driving gas temperatures is required if a range of test gas enthalpies is to be covered. In practice this limitation is often not severe, partly because the values of the stagnation enthalpy achievable under tailored conditions with unheated hydrogen and helium as driving gas correspond to values of full-scale interest, and partly because it appears that the secondary reflected shock is often too weak to have a serious effect on the flow duration, even when the conditions depart appreciably from those for "tailoring".

The operation of the "reflected-shock" shock tunnel is in many ways similar to that of the gun tunnel

*Beyond that obtainable by using a nozzle diaphragm and evacuating the working section as discussed previously.

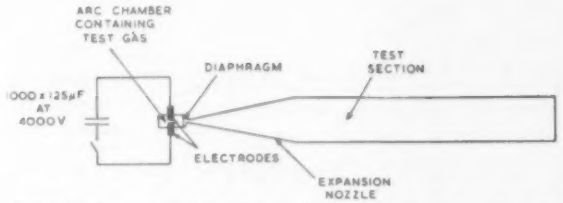


FIGURE 10. A "Hotshot" tunnel. (Electrical storage capacity is that of the 16in. facility at A.E.D.C.)

described in Section 3.2.1. In the former it is simpler to obtain very high test enthalpies because it is unnecessary to accelerate a piston, and it is also simpler to calculate the stagnation conditions because the compression occurs through only two shocks instead of in a multiple reflection process. On the other hand, the presence of the piston in the gun tunnel eliminates the reduction of flow duration associated with acceleration of the shock tube contact surface discussed in Section 3.2.2.

3.2.4. *"Hotshot" Tunnels*

Another scheme is to heat the test gas supply not by shock compression, but by an electrical discharge, and this is used in the "Hotshot" type of tunnel developed[20, 21] at the Arnold Engineering Development Centre, and sketched in Fig. 10. Here the test gas is initially contained in a comparatively small chamber separated by a diaphragm from the expansion nozzle and working section. The gas is heated at constant volume by discharging through it between electrodes a large quantity of electrical energy stored in a capacitor bank, in an inductance, or in the flywheel of a generator. The circuit values shown in Fig. 10 are for a comparatively small tunnel at AEDC, and for details of the larger facilities References 20 and 21 should be consulted. When the diaphragm bursts, this hot high pressure gas expands through the nozzle into the working section, and in this way high Mach number flows at high temperature may be achieved. One advantage of the device is that it gives a running time that is larger than for a shock tunnel of comparable size. Another advantage is that the Reynolds number and densities tend to be higher. The main uncertainties concern the uniformity and composition of the resultant flow, and the determination of the stagnation conditions, but it appears from References 20 and 21 that these uncertainties are not severe.

3.2.5. *Techniques using Moving Models*

The other techniques that are used to investigate hypersonic flows involve moving models propelled by rockets or launched from high velocity guns. By using as propellant a light gas heated by combustion, shock compression, or by an electrical discharge, it is possible[22, 23] to launch models from guns at velocities up to about 15,000 ft./sec. If such models are launched into a ballistic range, Mach numbers of about 15 may be obtained in still air. The Mach number may be further increased[22] by launching against an air stream as in the free-flight tunnel sketched in Fig. 11, and

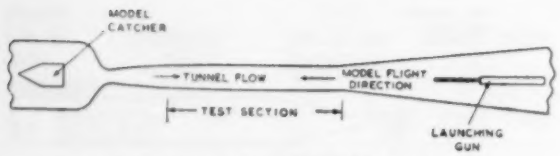


FIGURE 11. A free-flight wind tunnel.

developed at the Ames Research Centre. Here the model is launched from a gun, located in the diffuser of a wind tunnel, against the tunnel air stream. Free-flight techniques of these types provide powerful methods for simulating full-scale conditions, although, of course, the amount of data that can be measured is limited.

Finally, in any survey of experimental methods one must not overlook free-flight rocket-propelled model tests[4]. These are likely to be expensive and to involve difficulties, but they will no doubt play a vital role, especially in providing data under full-scale conditions for comparisons with the more detailed results of laboratory experiments.

4. The Extent to which the Full-Scale Conditions can be Simulated by the Experimental Facilities

A full discussion of the extent to which these experimental facilities can simulate full-scale conditions is beyond the scope of this paper, and some general features only will be indicated. For this purpose it is sufficient to present the data in the form reproduced in Fig. 12. Here the main aerodynamic parameters are considered, and the column for each facility extends vertically over the range of altitude for which the value of the parameter for the full-scale ballistic missile considered previously may be simulated.

For simplicity certain fixed design pressures, temperatures and dimensions have been assumed for the facilities. These represent values that have either been achieved, or which could be achieved without major extensions of existing knowledge, and are as follows:

(*i*) The stagnation pressure in the tunnels using helium and heated air is 100 atmospheres absolute.

(*ii*) The stagnation temperature in the helium tunnel is 288°K, and that in the heated-air tunnel is 1,000°K or the temperature required to avoid liquefaction, whichever is the smaller.

(*iii*) The shock tube and shock tunnel are driven by unheated hydrogen at pressures up to 1,000 atmospheres absolute.

(*iv*) The maximum pressure in the "Hotshot" arc chamber is 1,000 atmospheres absolute, and the maximum electrical energy is 10^6 Joules.

(*v*) The maximum launching velocity in the ballistic range and free-flight tunnel is 15,000 ft./sec.

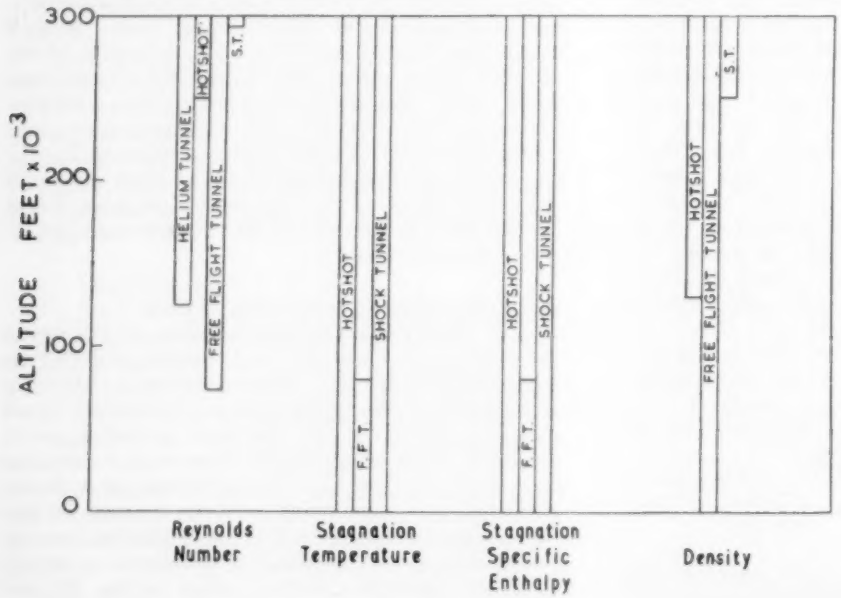
(*vi*) The maximum pressure in the ballistic range and the maximum stagnation pressure of the free-flight tunnel is 10 atmospheres absolute.

(*vii*) When the model is stationary the scale is 1/10 full size; when the model is moving the scale is 1/100 full size.

Figure 12 illustrates the simulation conditions if it is required to study a flow for which it is necessary to reproduce the high Mach number accurately, and it is drawn for $M = 20$. Of the facilities considered, only four are capable of giving this Mach number, and when looking at their performance we must remember that, for the ballistic missile, aerodynamic phenomena are of greatest interest below 100,000 ft.

We see that the helium tunnel gives a useful Reynolds number, and that the free-flight tunnel is even better. The Reynolds numbers of the "Hotshot", and especially of the shock tunnel, are very low.

The helium tunnel is, of course, quite unable to reproduce the full-scale stagnation temperature or enthalpy, and is accordingly not considered in the remaining columns. Full-scale temperature and enthalpy can be achieved in the "Hotshot" and the shock tunnel, and the free-flight tunnel is also satisfactory*. The free-flight tunnel and the "Hotshot" tunnel are



*It should be remembered that the presentation of Figs. 12-14 gives a pessimistic picture of certain of the simulation comparisons because, for example, temperature and enthalpy are almost constant above about 120,000 ft. (*see* Fig. 1).

FIGURE 12. Simulation of ballistic missile by facilities operating at $M = 20$.

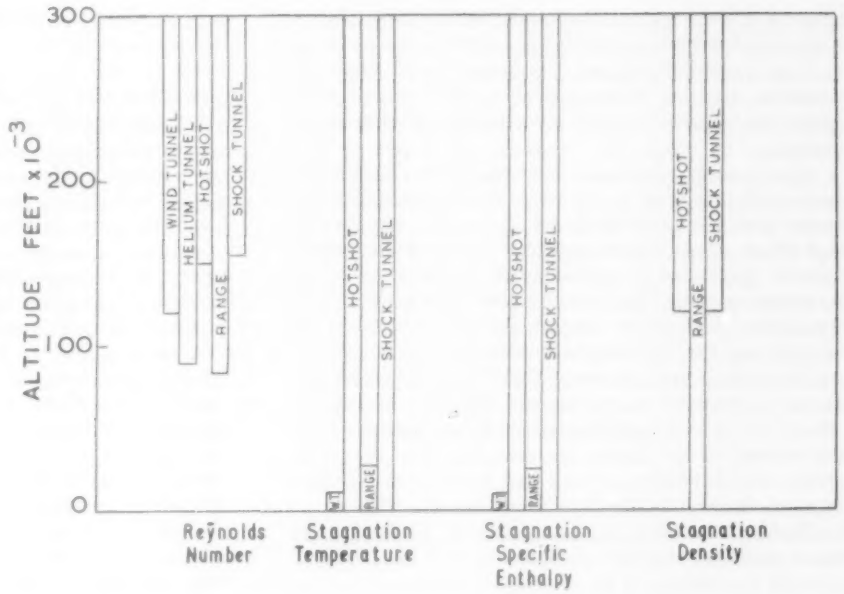


FIGURE 13. Simulation of ballistic missile by facilities operating at $M = 7$.

considerably better than the shock tunnel on the basis of density. The stream density is considered here rather than the stagnation density, since it is likely to be more appropriate in regions of the flow whose study requires fairly precise Mach number simulation.

Space does not permit a consideration of the simulation conditions for a glide vehicle, but reference to Figs. 1 and 2 indicates that the full-scale values of the parameters are broadly similar to those for the ballistic missile at altitudes between about 100,000 and 200,000 ft. It should, therefore, be simpler to simulate the conditions for the glide vehicle since this does not involve hypersonic flight at the relatively low altitudes of the ballistic missile.

Figure 13 illustrates an approach that is sometimes adopted to reduce the limitations on simulation indicated in Fig. 12. It is argued that, for many full-scale configurations, exact simulation of Mach number is unlikely to be necessary, provided that the Mach number of the model test is kept large. Accordingly, in Fig. 13 it is assumed that the facilities are operated at $M = 7$, and the simulation of the other parameters is reconsidered. The wind tunnel, using heated air, and the ballistic range now enter the field, whereas we no longer consider the free-flight tunnel as its value has already been demonstrated.

At the reduced Mach number it is seen that the Reynolds numbers and densities achieved in the "Hotshot" and shock tunnels are more realistic. Stagnation density is considered, since this is likely to be most representative in cases where departure from full-scale Mach number is permissible. The helium tunnel gives a Reynolds number substantially better than the tunnel using heated air, but both are rather low. The wind tunnel, helium tunnel and the ballistic range also give low values of the stagnation temperature and enthalpy. This comparison gives, of course, a grossly inadequate impression of the potentialities of the ballistic range, since in Fig. 13 the launching velocity has been restricted to about 7,000 ft./sec. in order to give $M = 7$, whereas values more than twice as large may be achieved. If this larger launching velocity is used, the Reynolds number, stagnation temperature and stagnation enthalpy lie close to the full-scale values.

Extending the argument leading to the reduction of Mach number, Fig. 14 shows the simulation conditions if the Mach number at which the facilities operate is reduced to 3. Here it is argued that simulation of Mach number is unnecessary, as may be the case if the experiment is on the flow near the stagnation

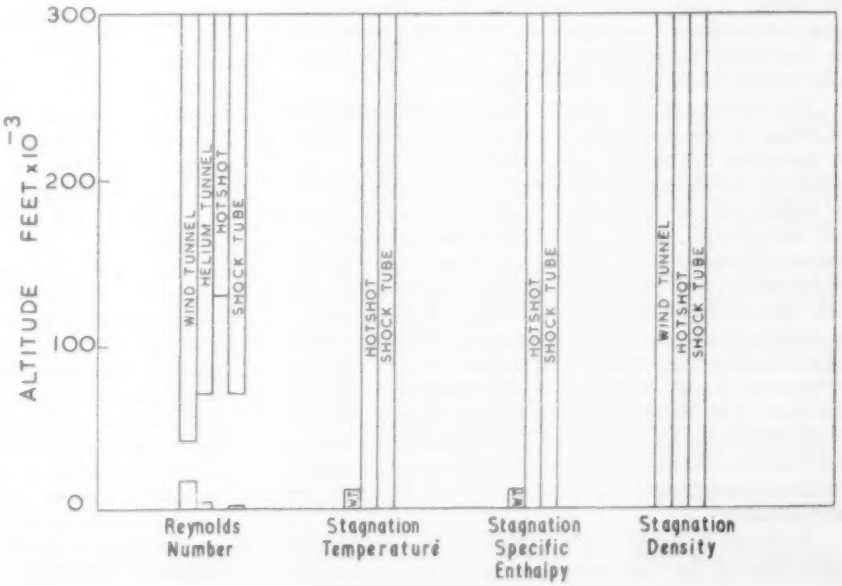


FIGURE 14. Simulation of ballistic missile by facilities operating at $M = 3$.

point of a bluff body. The unexpanded shock tube is now considered and is seen to give good simulation.

Gun tunnels have not been considered in these examples, because, when pushed to the limit of their performance, the test conditions resemble those in shock tunnels.

The general conclusion that may be drawn from studies of this type is that the range and free-flight tunnel probably offer the best full simulation of flow at high Mach numbers, although the "Hotshot" and shock tunnels give useful performance if relatively low Reynolds number and density are acceptable. The simulation problem is simpler for the glide type of vehicle than for the ballistic missile.

If, for fundamental work, high Mach number flows are to be studied in the absence of high-temperature effects, the helium tunnel appears to be a valuable tool. For studies of the hypersonic flow near the stagnation points of bluff bodies, it seems that several facilities may be used, the shock tube probably being the best.

Perhaps the major point to note is, however, that many types of facility enable useful extents of the relevant parameters to be covered in laboratory experiments, so that research may be conducted to determine the important ranges, or to confirm or construct theories which may make it unnecessary to simulate full-scale conditions fully. There is an obvious analogy with conventional wind tunnels which usually operate at Reynolds numbers that are low compared with the full-scale values, but, nevertheless, yield valuable results provided that the resultant scale effects are fully understood.

5. The Uniformity of the Test Stream

In addition to considerations of simulation, the value of an aerodynamic test facility also depends on the uniformity of the flow that it provides. Ideally the test stream should be of uniform undissociated, unionised air moving at high velocity relative to the model, so that when dissociation and ionisation occur they do so, as in free flight, under the influence of the temperature and density fields of the model itself.

The heated air and helium tunnels present no new problems in this connection, but in some of the other facilities the test stream is heated before it meets the model to a temperature sufficient to cause dissociation or ionisation. In some cases, such as the shock tube, the temperature of the stream immediately ahead of the model remains very high. In others, such as the shock tunnel, with expansion to large Mach number and low temperature immediately ahead of the model, recombination[24] or deionisation may not occur fully when the flow is cooled by expansion in a nozzle of practicable length, especially if the working pressures are low. The test stream approaching the model does not then correspond to that under free-flight conditions; the resultant effects are not fully understood and require further investigation, but they may not be large in certain cases.

It should be noted that uncertainties concerning the properties of the free-stream ahead of the model are largely absent in the ballistic range and the free-flight tunnel.

6. Measuring Techniques

The utility of an experimental facility also depends on the nature and number of measurements that can be made in the period of steady flow that it provides. This period will be equal to the duration of uniform flow provided by the facility minus the time taken to establish steady flow past the model under test. This latter time is at present largely unknown, and will probably depend on the geometry of the model, but a crude estimate suggests that it may be typically about 50 microseconds. In some facilities this represents an appreciable fraction of the running time, and the time left for making observations of the flow is thereby seriously reduced. Problems of techniques and instrumentation thus play an unusually important part in many hypersonic facilities. Moreover, at high working temperatures, additional quantities may need to be measured in order to understand the flow. This arises, for example, because in the presence of dissociation, three instead of two quantities need to be measured to define the state of the gas, and this number is further increased if ionisation is present.

In describing measuring techniques, no discussion will be given of the requirements of hypersonic wind tunnels using heated air or helium, because adaptations of conventional wind tunnel technique are usually adequate. The techniques used in ballistic ranges and free-flight tunnels will also not be discussed, because these are specialised and are described, for example, in References 22 and 23.

Measurements are required either for deriving the aerodynamic loading, forces or heat transfer experienced by the model, or for observations in the empty test stream aimed at assessing its uniformity and average properties.

For force measurements, suitably designed strain-gauge balances have response times suitable for use in gun and "Hotshot"[20] tunnels. For shorter operating times, like those of some shock tunnels, a number of accelerometer balances are in a comparatively early stage of development. An alternative technique[23] is to suspend the model by weak wires or threads which break when the flow strikes them. The subsequent acceleration of the model is then observed by high-speed photography, and the aerodynamic forces may then be deduced.

For pressure measurements, there are several transducers[10, 11, 25, 26] with response times suitable for gun and "Hotshot" tunnels, and also with the shorter response times[27, 28] required for shock tubes and tunnels. Such gauges are frequently small enough to be mounted within a model of reasonable size for the measurement of the pressure distribution at its surface. The sensitivity of a pressure transducer usually falls as its response time is reduced, so that, in facilities with very short running times, it may sometimes be difficult to measure low pressures accurately. Some transducers are also highly sensitive to acceleration, and must be mounted accordingly.

For heat transfer measurements, thin-film resistance thermometers[29, 30] are in a relatively advanced stage of development, and have response times that are short enough for use in any of the facilities described. Techniques also exist[20, 29, 31] for measuring larger rates of heat transfer than can be handled by thin films.

For flow pattern observations, schlieren methods, using short duration single[32] or multiple spark[33] light sources, play an important part. In some facilities difficulties may arise because the density may be low, and the sensitivity of a conventional schlieren system becomes inadequate, especially for boundary layer and wake studies. The phase-contrast schlieren method gives a considerable increase of sensitivity over the conventional methods, but even this is sometimes not sufficient. In some cases there is a need for a visualisation technique with a sensitivity of perhaps 100 times that of a conventional schlieren system, and a possible approach is to use the electron-optical schlieren system suggested by L. L. Marton, of the National Bureau of Standards[34]. A simple and useful technique in high enthalpy facilities is to take direct photographs of the visible radiation emitted by the high temperature gas near the model.

In short duration, high enthalpy facilities, a promising approach for measurement of the stream temperature is to use spectrographic techniques. A reversal technique[35] has been used with considerable success to temperatures of about 3,000°K, and several spectrographic techniques are being developed[36, 37, 38] for the measurement of higher temperatures. Optical interferometers may be used for density measurements[39], or, for lower densities, the absorption of an electron[40] or X-ray[41] beam may be employed. For ionisation measurements, the electron density, and other relevant quantities, may be determined by observations[42] of the absorption and reflection of microwaves from the hot gas, or by measurements[43] of its d.c. conductivity.

In addition to being useful in aerodynamic research, these and related techniques have been used with shock tubes for a wide range of research on problems of chemical kinetics and allied topics.

The velocity of the test stream may be inferred from measurements of other quantities, such as static and Pitot pressure (in the interpretation of Pitot pressure measurements, an allowance must sometimes be made for real gas effects occurring when the stream is brought to rest near the mouth of the Pitot tube). Alternatively, attempts are sometimes made to measure the velocity directly by, for example, finding photographically[10] the position of the disturbance from a spark discharged in the stream at a known time after the discharge took place. The Mach number of the test stream may also be derived indirectly from other measurements, or it may be estimated by measuring the shock angle from a wedge or cone by the use of schlieren photography.

7. Conclusion

It is hoped that this brief review has indicated that a wide range of experimental facilities and measuring techniques may be used for research on aerodynamic problems arising in hypersonic flight. In all cases there is a great need for further research and development work to explore the uniformity and physical properties of the flow produced or to extend the operating time, and to assess the accuracy and utility of measuring techniques and develop improved techniques. Much of this information can be obtained with relatively small and simple equipment, but some of it will only come from further experience with the use of the larger test facilities. Research is also required to indicate the important ranges of the relevant similarity parameters, so that future test facilities may be designed appropriately.

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A film showing the operation, mechanical arrangement, and instrumentation of a hypersonic shock tunnel at the N.P.L. was shown after the paper had been read.

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General Characteristics of Hypersonic Flow Fields

by

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SUMMARY: The object of this paper is to give a brief account of the development of the study of hypersonic flow over the past three decades, and to stress some of the ways in which hypersonic flow differs from supersonic flow. In some of the early work on the subject a wrong emphasis was given to the study of flows with attached shock waves and to the nature of the shock boundary layer interaction problem, and it is only quite recently, mainly as a result of experimental work, that it has become possible to obtain a balanced picture of the important features of hypersonic flow fields. In the present paper real gas effects are not taken into account; in general, the inclusion of these effects will not markedly alter the picture given.

Definition of Hypersonic Flow

A precise definition of what is meant by "hypersonic" is still a matter of some dispute among aerodynamicists; it is, however, generally agreed that Mach numbers above 5 qualify for the description[1]. A feeling for the rôle played by the Mach number may be obtained by considering the energy equation in the form:

$$\frac{u^2}{2}+\frac{a^2}{\gamma-1}=\text{constant}$$

where u is the local flow velocity,
a is the local speed of sound,
and γ is the ratio of specific heats.

This equation expresses the fact that the sum of the velocity energy/unit mass ($u^2/2$) and the temperature energy/unit mass ($a^2/(\gamma-1)$) is constant for the flowing gas. The ratio of the velocity energy to the temperature energy may then be expressed in terms of the Mach number, M_∞, as

$$\frac{\text{Velocity energy}}{\text{Temperature energy}}=\frac{\gamma-1}{2}M_\infty^2$$

and it is evident that for a Mach number of 5 the temperature energy has become small compared with the velocity energy. This is another way of saying that the effect of random collisions between the atoms or molecules of the flowing gas is of relatively little importance, and that the gas may be regarded to a first approximation as a stream of non-interacting particles. This is in fact the gas flow model which was proposed by Newton in 1687[2] and the concept was developed by Busemann (1933)[3] and von Kármán (1935)[4] for hypersonic flow.

Newtonian Theory

To determine the distribution of pressures over a body the assumption is made that the normal component of momentum is destroyed when the gas particles strike the surface, and that only the tangential velocity component, $M_\infty \sin\theta$ (where θ is the inclination of the body surface to the flow) remains: that is to say the flow becomes parallel to the surface of the body, a state of affairs which is a good approximation to the flow behind a shock wave over the forward part of a bluff body. From the Bernoulli equation, the pressure on the surface may be obtained and, assuming that the pressure at the shoulder falls to the ambient static pressure, (the so-called "modified Newtonian" theory), the pressure coefficient C_p, may be written

$$\frac{C_p}{C_{p\,\max}}=\sin^2\theta.$$

It has been found that this simple theory yields good agreement with experimental results over the forward part of a body at hypersonic velocities as illustrated by Fig. 1, which shows a measurement of the pressure distribution over the hemispherical portion of a hemisphere cylinder made at a Mach number of 9·8 in the

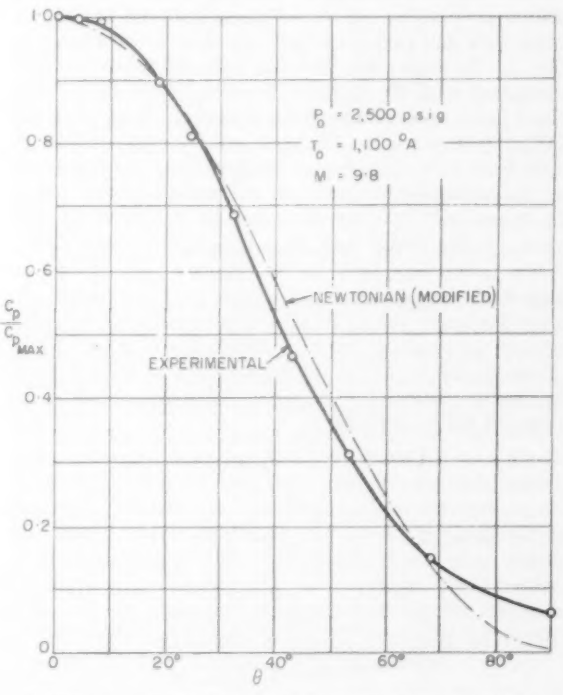


FIGURE 1. Pressure distribution over hemisphere cylinder.

Medium Hypersonic Gun Tunnel at the Armament Research and Development Establishment[5]. Towards the shoulder the measured pressures are higher than those given by Newtonian Theory, and it has been found that a better approximation may be obtained by matching the Newtonian pressure distribution over the front part of the body to the pressure distribution given by a Prandtl-Meyer expansion[6] over the rear part of the body. The matching point is taken as the point on the surface at which both the pressure and the pressure gradient are the same for the two distributions.

Flow Past Slender Bodies

The next development in hypersonic flow was a study of flows past slender bodies with attached shock waves. The perturbation equations for a slender body were developed by Tsien (1946)[7] using a method similar to that employed for supersonic flow. The main difference between the flow field considered for hypersonic flow and that for supersonic flow is illustrated by Fig. 2. In supersonic flow the body thickness is small compared with the distance between the body and the shock, and in developing the equations terms of order $(M_\infty \tau)^2$ (where τ is the thickness ratio) may be neglected. This leads to a considerable mathematical simplification as the supersonic perturbation equations are then linear. In hypersonic flow on the other hand, the shock lies close to the body, and $M_\infty \tau$ may be of order unity. Extra terms thus have to be retained and the set of hyperbolic equations which result are non-linear. It should however, be noted that Mach number is involved only in the combination $M_\infty \tau$; this leads to the so-called Tsien-Hayes similitude by which, for slender bodies, similarity exists for hypersonic flows having the same value of the product $M_\infty \tau$.

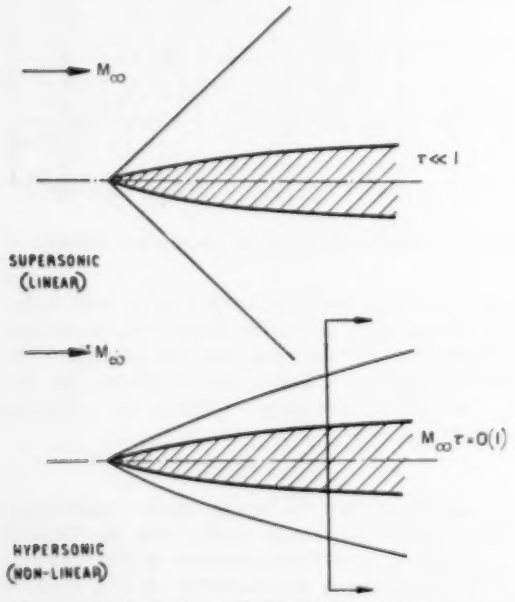


FIGURE 2. Small perturbation theory.

Flow Past Blunt Bodies

The results obtained using the hypersonic similarity theory agree neither with the experimental observation that aerodynamic coefficients tend to a constant value for high Mach number, nor with the results of more exact theory. The main reason for this, as pointed out by Oswatitsch[8] is that Tsien does not take account of the effect of the detached shock which must exist if the body thickness is not to become impossibly small as the Mach number increases.* In fact the case of practical interest for high Mach numbers is not the thin body with an attached shock, but a body for which the flow deflection angle is likely to be large compared with the Mach angle and which has a detached shock. Some of the important features of such a flow may be derived by considering the equations for an oblique shock at an inclination δ to the flow direction.

For the normal velocity and density changes across the shock,

$$\frac{\dot{u}_n}{u_\infty \sin \delta} = \frac{\rho_\infty}{\dot{\rho}} = 1 - \frac{2}{\gamma+1}\left(1 - \frac{1}{M_\infty^2 \sin^2 \delta}\right)$$

and for the pressure change,

$$\frac{\dot{p}}{p_\infty} = 1 + \frac{2\gamma}{\gamma+1}(M^2 \sin^2 \delta - 1).$$

(The circumflex ˙ refers to conditions downstream of the shock, and ∞ refers to free stream values).

Then for the $M_\infty^2 \sin^2 \delta \gg 1$, the velocity and density ratios become

$$\frac{\dot{u}_n}{u_\infty} = \frac{\gamma-1}{\gamma+1}\sin \delta \qquad \frac{\dot{\rho}}{\rho_\infty} = \frac{\gamma+1}{\gamma-1}$$

the pressure coefficient,

$$C_p = \frac{\dot{p} - p_\infty}{\frac{1}{2}\rho_\infty u_\infty^2} = \frac{4}{\gamma+1}\sin^2 \delta$$

and the speed of sound ratio

$$\frac{\dot{a}}{u_\infty} = \frac{\sqrt{\{2\gamma(\gamma-1)\}}}{\gamma+1}\sin \delta.$$

Thus the flow parameters, such as velocity, density and also the Mach angles and inclination of the streamlines behind the shock depend only on the flow inclination and not on the value of the Mach number for flow at sufficiently high Mach numbers. This is called the "Flow Field Freeze Principle" and applies over most of the forward facing portion of a typical body. Even at only moderately high supersonic velocities the tendency towards the hypersonic condition in the flow past a bluff body is quite marked, and above a Mach number of 5 such quantities as the shock shape and pressure coefficients have almost achieved their frozen values.

*There appears, however, to be a range of blunt cones and wedges with semi-angles between 10° and 50° approximately which have an attached shock, and for which, because of the relatively high pressure over the body, boundary layer interaction effects are small. It has been found that for these bodies at hypersonic speeds the pressures on the surface may be predicted with fair accuracy by using the methods of Ref. 27.

FIGURE 3(*a*). Flow past a wedge at $M = 3{\cdot}5$.

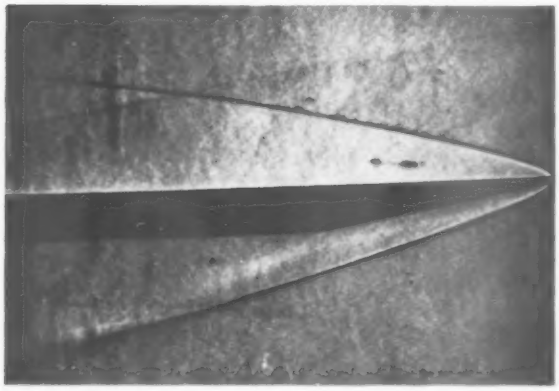


FIGURE 3(*b*). Flow past a wedge at $M = 9{\cdot}8$.

Fig. 3 shows the flow past a blunt-nosed wedge for two Mach numbers, 3·5 and 9·8. Already at $M = 3{\cdot}5$ the curvature of the shock is noticeable for a large distance back from the leading edge of the wedge and, except for the slope very far back from the leading edge, the shock shape shown for $M = 9{\cdot}8$ remains effectively unchanged for Mach numbers above 5.

Blast Wave Analogy

Another approach to hypersonic flow was suggested by Hayes[9]. The hypersonic perturbation equations have the same mathematical form as those for unsteady flow in one less space dimension. Thus the flow in the x-direction past a two-dimensional body is regarded as being equivalent to the transient flow as viewed by an observer at rest in the y, z plane produced by the passage of a body. Alternatively the flow may be

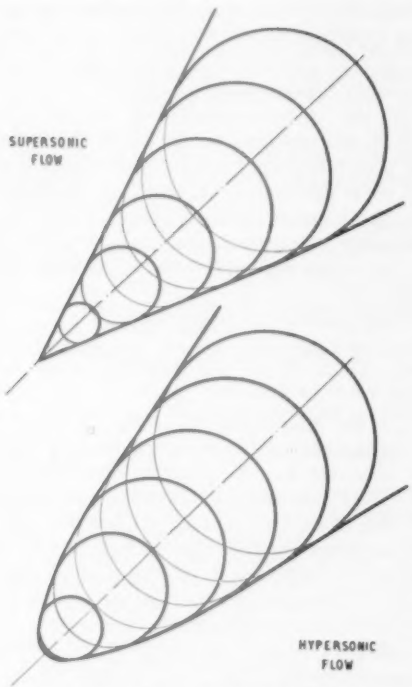


FIGURE 4. Generation of shock by cylindrical waves.

regarded as that produced by the expansion of two opposed pistons moving along the body with the free stream velocity, u_∞ (Fig. 2). Similarly in three dimensions the flow may be thought of as that generated by an expanding cylindrical piston (or line charge). The resulting shock wave configuration is shown in Fig. 4. The upper picture shows the familiar generation of a shock in supersonic flow by weak supersonic cylindrical waves generated near the axis, whereas the lower picture shows the shock shape generated by the strong cylindrical waves associated with hypersonic flow. (The shock envelope is a paraboloid.) This concept, the so-called "Blast Wave Analogy," enables one to use the methods of the considerable body of theory developed for strong blast waves from a constant energy source[10, 11, 12].

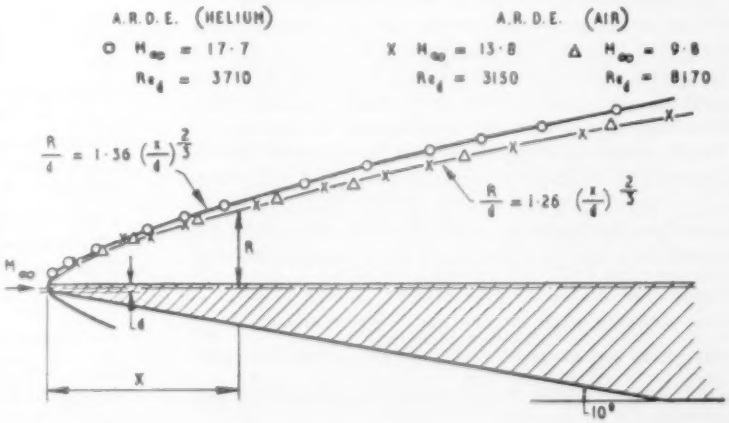


FIGURE 5. Shock wave shape for flat plate with blunt leading edge ($d = 14{\cdot}2 \times 10^{-3}$ in.).

For instance for two-dimensional flow we have, for the plane explosion (Lees[137])

$$R(t) = K_0(\gamma)\left(\frac{E}{\rho_\infty}\right)^{1/3} t^{2/3}$$

where R is the distance of the shock front from the origin at time t, E is the energy of the explosion per unit time and ρ_∞ is the ambient density. K_0 is a function of γ which has a value close to unity. Using the Blast Wave Analogy, we replace the time t by x/u, and we identify the energy per unit length, E, with the total drag of the body, D.

Then the shock shape is given by

$$\frac{R(x)}{d} = K_1(\gamma) C_d^{1/3}\left(\frac{x}{d}\right)^{2/3}$$

where C_d is the drag coefficient, $D/\frac{1}{2}\rho_\infty u^2 d$, and d is the leading edge dimension. A corresponding expression may be obtained for the pressure distribution on the surface. Fig. 5 shows typical shock shapes obtained in the A.R.D.E. 3 in. Gun Tunnel for the flow past a flat plate with a blunt leading edge, at high Mach numbers. Apart from the region close to the leading edge, good agreement is obtained between the experimental shock shapes and those predicted by blast-wave analogy. Strictly speaking, as has been shown by Lees[13] the blast wave analogy applies only to a class of bodies of the form $y \simeq x^m$.* This includes the flat plate and cylindrical afterbodies, but not a blunted wedge or cone. A recent paper by Cheng[14] has considered an extension of the blast wave analogy to blunt nosed bodies with slender afterbodies.

Shock Boundary Layer Interaction

Soon after Tsien had published his paper on hypersonic similitude, he noted that because the shock lay close to the body his conclusions would probably be invalidated by interaction effects between the shock and the boundary layer. It was also found experimentally that the pressure over a flat plate, instead of being constant as would be expected by Tsien's theory, falls off with distance from the leading edge. A number of authors studied this problem theoretically[15]-[19] and it was concluded that there would be an important effect of the boundary layer growth on the external flow. A number of important regions of shock boundary layer interaction may be distinguished (Fig. 6): a *leading edge region* where it is probable that the shock is always detached, and where, for very small leading edge dimensions slip flow effects may become of importance, a *strong interaction region*, where the external flow is influenced by the boundary layer growth and vice versa, and a *weak interaction region* farther back along the body where the boundary layer growth is affected by the external flow but the influence of the boundary layer on

*Flow similarity exists for $m' \leqslant m \leqslant 1$, where $m' = 3/4$ for a planar body and where $m' = \frac{3}{2}\left(\frac{\gamma+1}{3\gamma+2}\right)$ for a body of revolution. For $m < m'$ the flow field will depend on the nose geometry.

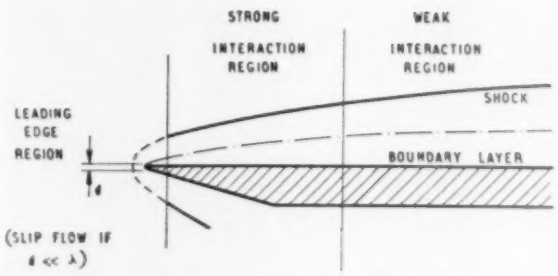


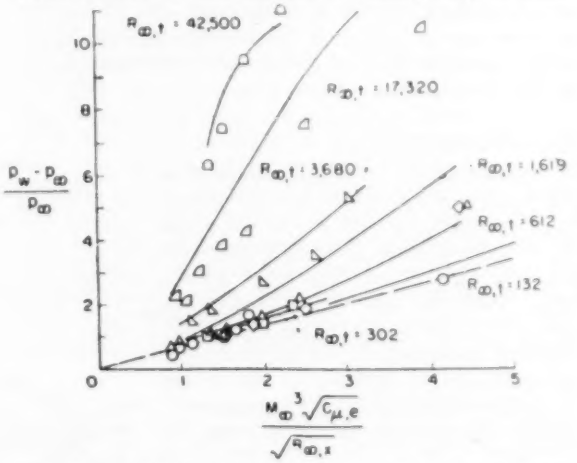
FIGURE 6. Shock boundary layer interaction.

that flow is considered to be negligible. Lees and others showed that for both these interaction regions the governing viscous interaction parameter (for the flow past wedges which he considered) is

$$\frac{M_\infty^3 \sqrt{c_{\mu,e}}}{\sqrt{R_{\infty,x}}}$$

(where $R_{\infty,x}$ is the Reynolds number based on the free stream conditions and the distance x from the leading edge of the plate and $c_{\mu,e} = \mu_e T_\infty / \mu_\infty T_e$ where μ_e is the dynamic viscosity at the recovery temperature T_e, and μ_∞ is the dynamic viscosity at the free stream temperature T_∞).

Measurements such as those of Bertram[20] and Kendall[21] seemed to confirm the theoretical predictions, whereas measurements by Bogdonoff[22] in the Princeton Helium Hypersonic Tunnel at Mach numbers from 11 to 14 showed poor agreement with the theory. It was not until about 1955 that it was generally recognised that there were two separate effects influencing the induced pressures measured over flat plates and that, in addition to the viscous interaction there was at hypersonic speeds likely to be a strong inviscid effect due to the influence of the finite leading edge thickness. (It is impossible from an engineering point of view to manufacture a leading edge of much



—— linear addition of viscous and blunt leading edge theory.
— — viscous theory.

FIGURE 7. Combined viscous and leading edge effects on flat plate pressures ($M_\infty = 11{\cdot}4$).

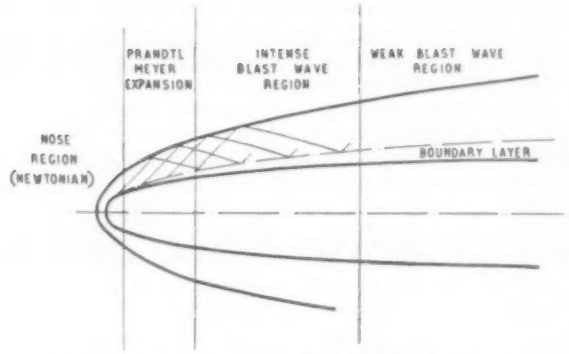


FIGURE 8. Blunt slender body in hypersonic flow.

less than 0·001 in. thick.) In a paper given in 1956, Lees[23] calculated the ratio of the energy imparted to the flow by the traverse field associated with the blunted nose to that due to the boundary layer growth and showed that the effects of blunting would be important for hundreds of leading edge thicknesses back from the leading edge. The relative importance of viscous and inviscid effects is dependent on the leading edge Reynolds number, $R_{\infty,t}$ and Bogdonoff[24] showed experimentally for flat plates that the effect of the leading edge dominated the flow for approximately $R_{\infty,t} > 1{,}000$. The induced pressure on a flat plate may be expressed as a linear addition of viscous interaction and inviscid effects; this is illustrated by Fig. 7 (taken from Ref. 25) in which the inviscid pressure difference at the wall, $P_w - P_\infty$, calculated by using a sonic wedge characteristics theory modified according to blast wave theory, has been added to the viscous effects. Using this method good agreement is obtained between theory and experiment over a wide range of values of leading edge thickness. The effects of nose blunting have also been considered for the flow over cones at hypersonic speeds by Bertram[25] and Bogdonoff[26].

The Blunt Slender Body

Since 1956 most of the theoretical effort in the hypersonic field has concentrated on calculating the inviscid flow field on blunt nosed slender bodies and, on the experimental side, to investigating heat transfer and the effects of yaw. Some of this work is dealt with in subsequent papers, but in concluding this general picture of hypersonic flow the methods used for determining approximately the pressure distribution over a typical blunt slender body such as that shown in Fig. 8 will be summarised. In the nose region the modified Newtonian pressure distribution followed by a Prandtl-Meyer expansion to the shoulder will give good results, and behind the shoulder the Blast Wave Analogy may be used to predict the surface pressures which may be expected.

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Some Special Aspects of Hypersonic Flow Fields

by

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1. Introduction

When a body moves through air at very high speed at such a height that the air can be considered as a continuum, the distinction between sharp and blunt noses with their attached or detached bow shocks loses its significance, since, in practical cases, the bow wave is always detached and fairly strong. In practice, all bodies behave as blunt shapes with a smaller or larger subsonic region near the nose where the entropy and the corresponding loss of total head change from streamline to streamline due to the curvature of the bow shock. These entropy gradients determine the behaviour of the hypersonic flow fields to a large extent. Even in regions where viscosity effects are small they give rise to gradients of the velocity and shear layers with a lower velocity and a higher entropy near the surface than would occur in their absence. Thus one can expect to gain some relief in the heating problems arising on the surface of the body. On the other hand, one would lose farther downstream on long slender shapes as more and more air of lower entropy is entrained into the boundary layer so that the heat transfer to the surface goes up again. Both these flow regions will be discussed here for the simple case of a body of axial symmetry at zero incidence. Finally, some remarks on the flow field past a lifting body will be made. Recently, a great deal of information on these subjects has appeared in a number of reviewing papers[1, 2] so that little can be added. The numerical results on the subsonic flow regions in Section 2 have not been published before.

2. Flow Field Between the Bow Wave and the Blunt Body

2.1. General Considerations

A great deal of both theoretical and experimental work has been done on the flow between the surface of a blunt body and a curved bow wave which forms ahead of it. Pressures and heat transfer on the surface have been measured and the shock shape investigated by schlieren. Since the temperatures behind the shock are high enough for the air to dissociate, the flow field depends on a large number of parameters (*e.g.* Reynolds number, Mach number, equation of state of the air at very high temperatures, relaxation time for dissociation and recombination) and theoretical predictions of their effects are very useful. For large Reynolds numbers viscosity effects are confined to the boundary layer, and inviscid flow theory with proper allowance for the entropy gradients predicts the flow field (shock shapes, pressure distributions) fairly well, provided that the proper equation of state is used. At high temperature the vibrational degrees of freedom in the molecules are excited, dissociation of the molecules N_2 and O_2 and chemical reactions take place, so that the ratio γ of enthalpy to internal energy decreases. Shock tube experiments appear to show that the air behind a strong bow wave is in equilibrium, *i.e.* the relaxation time for all these reactions is very short, so that equilibrium flow can be assumed as established behind the shock in the subsonic region. A Mollier diagram giving the pressure p as a function of the enthalpy h and the entropy s can be used, but because of its complexity this would require the use of electronic computers. Nevertheless, one can obtain some fairly good ideas about the behaviour of the flow in this region when assuming that γ is approximately constant throughout the subsonic region.

Even with this simplification of the equation of state the inviscid flow between the bow wave and blunt body represents a difficult mathematical problem. A number of attempts have been made to obtain analytic solutions by starting from the limiting case of very large Mach number and very small $(\gamma-1)$[3-9]. By a very simple argument Hayes[8, 9] showed for a spherical body in the limiting case of small M_∞^{-2} and small $\delta=\dfrac{\gamma-1}{\gamma+1}$ where the ratio of the density in front of the normal shock to the density behind approaches δ, that the flow between the body and the shock is of constant density, that the stand-off distance of the shock is $\delta\times$ body radius at nose, and that the velocity is constant along streamlines. The bow shock is concentric to the body and the tangential velocity component varies linearly between body and shock. By taking the change of velocity along the streamlines into account Hayes arrived at the following simple results for the ratio of the stand-off distance d_1 to the body radius R_0 and the distribution of the tangential velocity u:

$d_1/R_0=\delta\,(1+\sqrt{(2\delta)})^{-1}$, $u/u_s=\sqrt{(2\delta)}+(1-\sqrt{(2\delta)})\,y/d_1$

(u_s = tangential velocity at the shock, y = distance from

surface). The pressure along the shock is proportional to $\sin^2\alpha_s$ (α_s = slope of shock) and should be less on the surface due to the centrifugal forces (Busemann[3], Ivey, Klunker and Bowen[10]). Further various estimates of this effect lead to pressures on the surface, which are too low, and to zero pressure as the surface slope θ decreases from 90° to 30° along the sphere. Experiments and numerical solutions of the complete equation show that this singularity does not arise. In fact, the so-called modified Newtonian formula (*i.e.* adjusted to obtain the correct stagnation pressure) $C_p/C_{p\max} = \sin^2\theta$ gives fair agreement with experiments. The explanation can be found in the fact that shock and body are not concentric. The streamline curvature decreases with increasing distance from the surface which reduces the radial pressure drop along the radius and this drop is balanced by the higher pressure behind the less curved shock.

Hayes[8, 9] also developed the main ideas of the mechanism of the flow past flat-faced bodies. By utilising the same approximations as for the spherical nose, he derived an integral equation for the shock shape in terms of the body shape. The equivalent differential equation has a singularity at $\pi/2 - \alpha_s = \sqrt{(\rho_\infty/\rho_s)} = \sqrt{\delta}$ (α_s shock slope) and this point is taken to correspond to "choking" conditions at the shoulder. The solution of the equation cannot be continued from the flat face of the body round the shoulder unless this condition (which is independent of the body shape) is satisfied.

This points to the importance of the sonic region. For blunt faced bodies the radius of curvature near the shoulder is really the determining factor of the flow pattern as experiments for a series of round-nosed models have shown. (D. G. Fraasa and E. M. Wisenbaker, reported in Ref. 1). Fig. 1 shows the measured variation of the stand-off distance at the nose d_1 and at the shoulder d_2 with the nose radius R_0 for a Mach number $M = 5{\cdot}8$ (D = afterbody diameter). The sonic "choking" régime extends from $R_0/D = 1{\cdot}3$ to ∞. The ratio d_2/D is constant in this region, whereas the stand-off distance from the nose increases slowly with the nose radius. This régime corresponds to an inclination θ_{SH} of the shoulder of the body with respect to its axis of 67° to 90°. In the former case the shock layer at the shoulder forms almost a parallel-walled channel at the shoulder. For θ_{SH} between 45° and 67° the sonic point on the shock moves towards the axis with decreasing nose radius, whereas the sonic point at the shoulder remains fixed. For small values of θ_{SH} both sonic points move towards the axis and the stand-off distance is almost a linear function of the nose radius (d_1/R_0 = constant). Fig. 2 shows some measured pressure distributions plotted against distance along the surface referred to distance s^* to sonic point. The body shape has a large effect on the surface pressures and also on the velocity gradient at the forward stagnation point. The so-called Newtonian formula $C_p/C_{p\max} = \sin^2\theta$ fits fairly well for the spherical nose between stagnation point and sonic line and for bodies having small changes of curvature, but not so well for the other shapes. If the shoulder of a flat-faced body (radius of curvature at

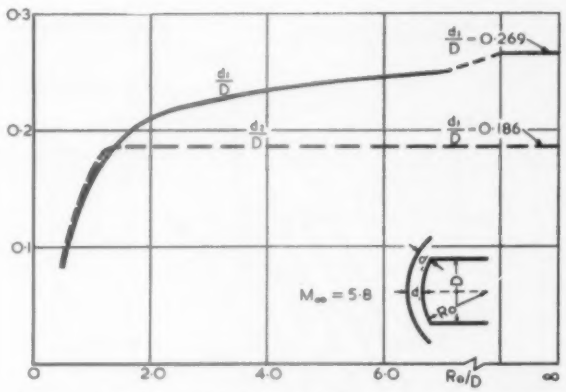

FIGURE 1. Shock stand-off distance plotted against nose radius. Experiments at $M = 5{\cdot}8$.

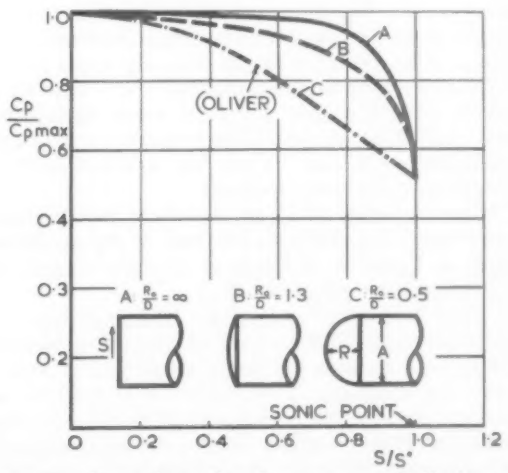

FIGURE 2. Comparison of surface pressure distributions on several blunt-nosed bodies, $M = 5{\cdot}8$.

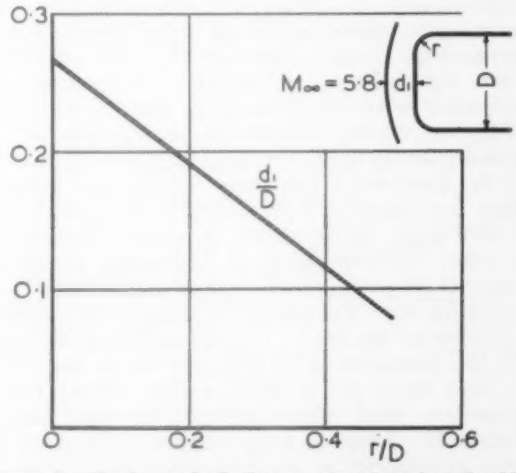

FIGURE 3. Shock stand-off distance plotted against shoulder radius, $M = 5{\cdot}8$.

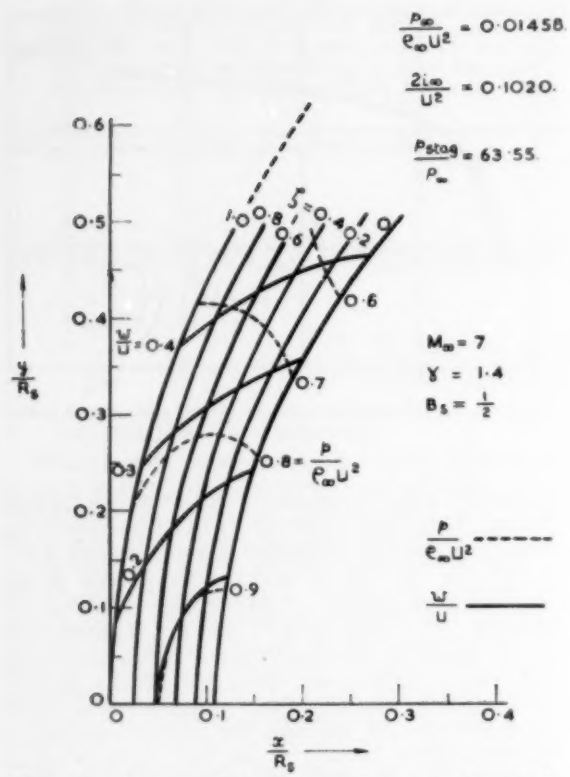

FIGURE 4. Axisymmetrical non-viscous flow past a blunt body (sphere). Lines of constant pressure and velocity.

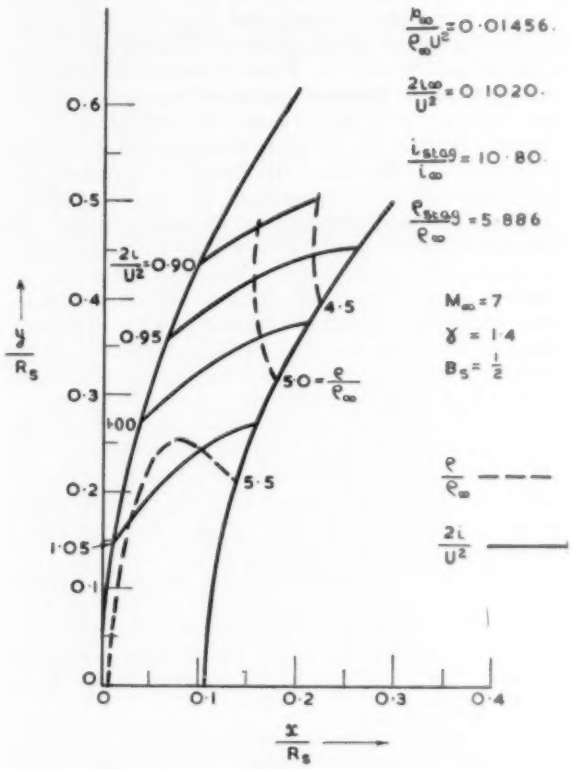

FIGURE 5. Lines of constant enthalpy and density.

nose ∞) is rounded, the stand-off distance at the nose decreases linearly with an increase of the shoulder radius (Fig. 3).

2.2. NUMERICAL RESULTS

Recently, several numerical methods have been developed for the analysis of the flow both in the subsonic and transonic regions in the U.S.A. and in this country (*see* Refs. 1, 2, 12, 13). Most of these methods consider the inverse problem of finding the body for a given shock shape. This simplifies the calculation, since the variation of entropy from streamline to streamline is known, if the shock shape is prescribed. With two velocity components, density, entropy and pressure known along the shock of a prescribed shape by the usual shock conditions, the first derivatives of these five quantities away from the shock can be calculated from the equations of motion (conservation of mass, two components of momentum, energy and equation of state). These can be used to obtain the values of these five quantities along a curve behind, but close to, the shock. Thus a "marching" technique for the calculation of the flow field can be established. A body shape (*i.e.* a stream surface starting from a stagnation point) which produces the shock can be calculated. Some of these numerical techniques are very sophisticated to avoid numerical instabilities. At R.A.E. Chester's independent variables have been used and suitable functions of the physical quantities which vary slowly as dependent variables. No difficulties as regards numerical instability have been encountered. Then the direct problem of finding the shock for a given body can be solved approximately by interpolation, if a sufficient number of solutions for variable shock shapes is available. If analytic shock shapes are assumed, the subsonic and supersonic part of the flow can be calculated together, without special consideration of the sonic line. For flat-faced bodies with a more rapid curvature near the sonic line the shape of the latter is very important. R. Vaglio-Laurin and A. Ferri[12] suggest a method which starts from a prescribed shock in the elliptic (subsonic) region and a prescribed sonic line. Then the subsonic region is calculated by difference methods.

The continuation of the flow field into the supersonic region can either be obtained by the above technique or better by the method of characteristics with the body shape prescribed.

The "marching" technique described was programmed for DEUCE and calculations were performed using conic sections for shock shapes. The resulting body shapes can also be represented with good accuracy as conic sections. It is easy now to relate body shapes and shock shapes with the aid of two parameters (radius of curvature at nose and "bluntness" parameter). Figs. 4 and 5 show part of a flow field ahead of a sphere

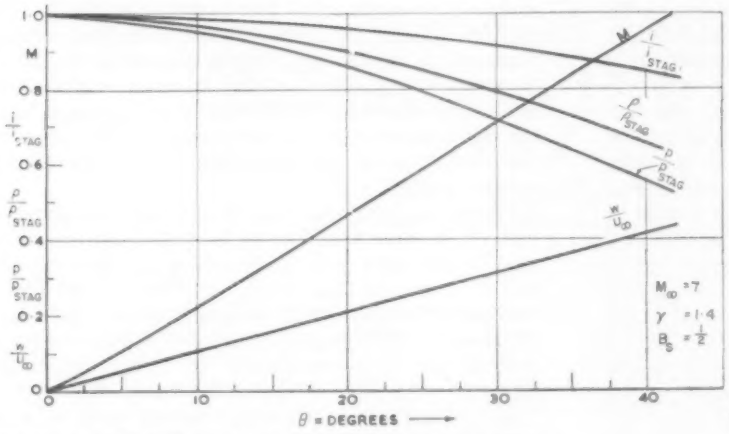

FIGURE 6. Distribution of velocity, Mach number, pressure density and enthalpy along the surface of a sphere.

at $M = 7$ ($\gamma = 1{\cdot}4$). Fig. 4 shows the isobars and the lines of constant velocity, and Fig. 5 the lines of constant density and enthalpy (temperature). Fig. 6 shows pressure velocity and Mach number distribution along the surface of the sphere. The so-called Newtonian distribution $\sin^2\theta$ is fairly close to the calculated one; the velocity varies almost linearly along the wall.

Figure 7 shows the stand-off distance referred to the radius of curvature of the shock at the nose divided by the density ratio across the shock plotted against γ. Results for various values of M_∞ and γ collapse fairly well to Hayes' approximate formula. The figure also shows the singular behaviour of the distance for small values of the density ratio. Unfortunately, the limiting case $\gamma \longrightarrow 1$ is only a mathematical limit which cannot be reached in practice. This becomes even more apparent if the velocity gradient along the surface is plotted against γ (Fig. 8). Again, the results collapse fairly well, but the velocity gradient is in practice always finite. When one considers high order derivatives like the second derivative at the stagnation point then the details of the body shape have an effect (Fig. 9). This derivative increases as the body shape ($y^2 = 2Rx - Bx^2$) varies from a hyperboloid ($B<0$) to a paraboloid

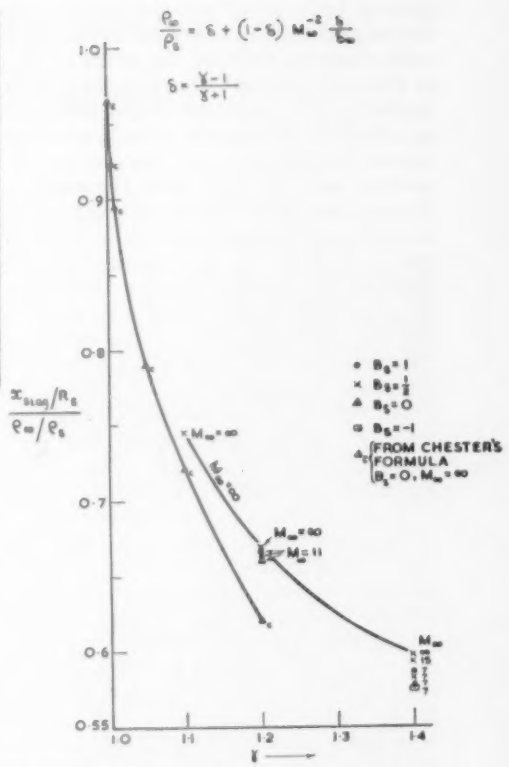

FIGURE 7. Variation of stand-off distance x_{stag} with γ, B_s and M_∞.

($B=0$), or sphere ($B=1$). The Newtonian value 2 fits best for a paraboloid.

3. Boundary Layer Behind the Shoulder of the Body

Along the forward-facing surface of the body the boundary layer thickness does not vary a great deal so that non-viscous theory predicts the flow field fairly well. After turning round the shoulder of a slender shape (*e.g.* a blunted cone or a semisphere cylinder) the boundary layer is subject to a falling pressure, the

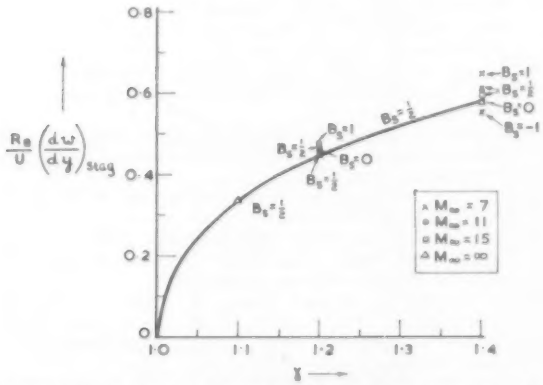

FIGURE 8. Variation of velocity slope at the stagnation point with γ.

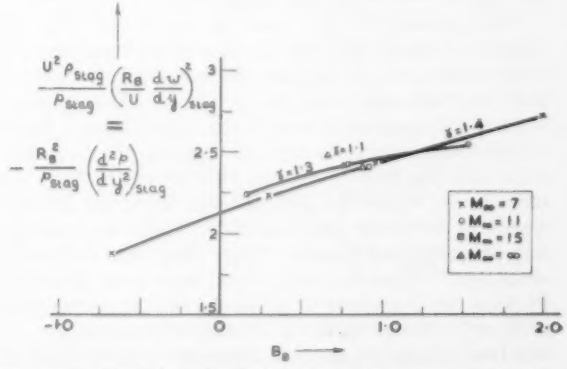

FIGURE 9. Variation of velocity slope and pressure at the stagnation point with γ, B_s, and M_∞.

gradient being larger near the shoulder and decreasing further downstream. Because of the shear layer which exists alongside the body, due to the strong curved shock at the nose, classical boundary layer theory cannot be applied directly even with proper allowance for the real gas effects (in the equation of state and in calculating the viscosity coefficient), since the entropy of the particles entering the boundary layer decreases as one moves downstream. Since the entropy decreases, the velocity of these particles rises faster in the streamwise direction than it would for the same distribution along the wall in the absence of entropy changes. Boundary layer theory is valid as long as the pressure gradients normal to the wall are small. To allow for this entropy effect the best method (suggested by Ferri[2]) appears to be to join the tangential velocity inside the boundary layer to the actual velocity of the entering particles and not to the velocity obtained from inviscid flow theory along the wall ($-dp/\rho = wdw + \mathrm{T}ds$). This requires that a boundary layer thickness is defined. On the other hand it avoids introducing a second boundary condition for the boundary layer at the outer edge for the normal derivative of the tangential velocity which could only be satisfied if the full Navier-Stokes equations were solved. For the case of constant pressure Glauert[11] suggested an interesting alternative where the proper joining up of the velocities is effected by a suitably defined velocity distribution at the start of the boundary layer. The effect of the decrease of entropy is a rise of the friction at the wall and a rise of the heat transfer to the wall.

4. Bodies at Incidence

If a body is yawed or cambered a side force or lift is produced. Its magnitude can be obtained by integrating pressures and shear stresses over the body surface. The well-known general momentum theorem, expressing the lift in terms of an integral of the pressures and induced velocities over a large control surface, is also valid for the hypersonic speed range. This control surface can be chosen in such a way that the lift appears in terms of a downwash integral over the Trefftz plane far behind the body provided that care is taken of the intersection of the bow shock and trailing edge shock with this plane. Unfortunately, the evaluation of this integral is impossible without further simplifying assumptions.

The creation of lift must be connected with the creation of vortex sheets in the flow as we know them at lower speeds, *i.e.* of surfaces across which in inviscid flow the tangential velocity would be discontinuous, whereas the pressure is continuous across them. These vortex sheets may arise from the trailing edges or from steps near the base where the flow separates from the surface, *i.e.* where the gas particles leave the surface. Again, as for lower Mach number, bubble-type separation probably will produce higher drag than the vortex sheet type. Thus the latter would have to be aimed at. Perhaps one may have to go all the way to obtain what is known at lower speeds as leading edge separation. In this case one would produce controlled vortex sheets all along lifting bodies by edges or steps.

The effects of these vortex sheets cannot easily be estimated since potential flow theory is not valid. On the other hand, the vorticity created by the entropy variations is akin to the vorticity created by friction in boundary layers and is confined to certain regions near the body or the wake. It is probably of little importance for the creation of lift, so that certain simplifications in the evaluation of the downwash integral for the lift appear possible.

Attempts have been made to calculate the perturbation of the flow field due to a small incidence of an axially symmetric body by linearisation with respect to this incidence. Vaglio-Laurin and Ferri[12] describe a numerical method which is based on a "marching" technique. According to this theory, the lift would be proportional to the incidence. In practice all bodies flying at very high Mach number are likely to be "slender," *i.e.* they will be inside the bow shock from the nose. Experiments[2] have shown that the Newtonian formula combined with a Prandtl-Meyer (two-dimensional) expansion can give a reasonable, if crude, estimate of the pressure distribution and the lift, provided that the curvature varies slowly along the surface. For the rear part of long slender shapes the results for cones at incidence can serve as asymptotic values for the pressure at the rear end, into which the distribution for the front end can be faired (Ferri). For spherical heads (*e.g.* semisphere cylinders) the subsonic region may be unaffected by incidence, the changes taking place in the supersonic region only. For other shapes, the movement of the forward stagnation point with incidence is not known, which makes the calculations of the flow field even more complicated.

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Hypersonic Boundary Layers with Special Reference to Heat Transfer

by

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1. Introduction

On the face of it boundary layer problems at hypersonic speed present enormous difficulties. However some aspects have turned out to be somewhat simpler than was visualised some years ago, and with the aid of certain simplifying assumptions many problems have been solved, at least to a good design approximation, and inaccuracies in the results are due not so much to the approximations involved, but to our lack of knowledge of the physical properties of air such as viscosity and Lewis number.

With the future emphasis likely to be on lifting bodies, at any rate for sustained flight at hypervelocities, we will start with a consideration of heat transfer to slender shapes suitable for such application.

2. Slender Bodies

It has been said that there is no such thing as a slender body at hypersonic speeds. Certainly the nose must be blunted to some extent to avoid excessive heat transfer rates and also to provide extra material to conduct away the heat. Aerodynamically, too, it has been demonstrated, notably by Bogdonoff and his co-workers, that the leading edge has a very strong influence on the flow field for large distances downstream, and thus on the heat transfer to the surface.

As an example of this leading edge influence we may take the results of Creager[1] who studied the effect of a hemicylindrical leading edge blunting on the local heat transfer and pressure distribution on a flat plate at a nominal Mach number of 4. There are actually two separate effects here; the first is the pressure gradient induced by the thick boundary layer growth even on a sharp-edged plate, and the second is the pressure gradient induced by a geometrical blunting of the leading edge. In general it is not possible to separate the two effects in a practical case, but we may note that a favourable pressure gradient reduces the heat transfer whereas the high local pressure produced near the leading edge increases the heat transfer. Amazingly, and rather fortunately, Creager found that his heat transfer measurements could be correlated by the simple theoretical flat plate formula

$$\mathrm{St} = \frac{h}{\rho u c_p} = 0{\cdot}332 P_r^{-2/3}\,\mathrm{Re}_x^{-1/2}$$

provided true local conditions are used in the calculations. Fig. 1 shows how this compares with experiment, assuming a Prandtl number of 0·72.

Now the non-linear variation with temperature of specific heat and viscosity of air can be accounted for by the intermediate enthalpy method (to be discussed later). This has been combined with the above formula and the results applied by Creager to the case of a flat plate with a leading edge diameter of one in. at a Mach number of 10 and free stream Reynolds number of one million, as shown in Fig. 2. This demonstrates that the blunting increases the heat transfer coefficient near the leading edge where high *static* pressures exist. The heat transfer is reduced however farther back along the plate where the static surface pressure approaches the free-stream value and the boundary layer induced favourable pressure gradient becomes the dominant influence. $M = 10$ is probably the limiting Mach number for this theory. Above this the entropy gradients produced by the curved shock influence the flow field, as discussed by Mangler.

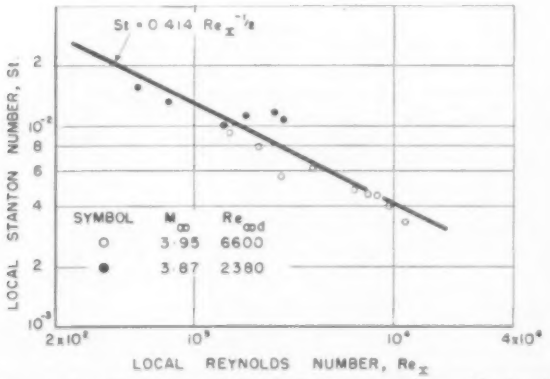


FIGURE 1. Variation of local Stanton number with local Reynolds number for a cylindrically blunted flat plate.

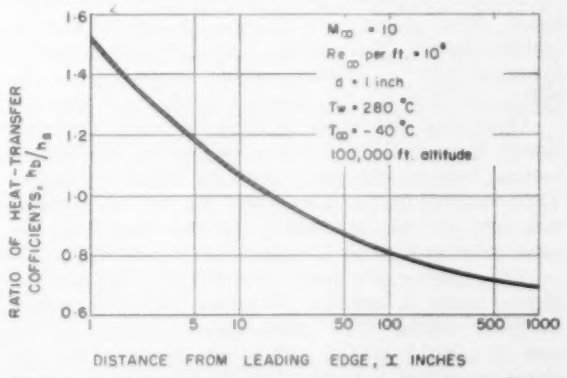

M$_\infty$ = 10
Re$_\infty$ per ft. = 10^6
d = 1 inch
Tw = 280 °C
T$_\infty$ = - 40 °C
100,000 ft. altitude
RATIO OF HEAT-TRANSFER COFFICIENTS, h$_b$/h$_s$
DISTANCE FROM LEADING EDGE, x INCHES

FIGURE 2. Effect of blunting on the heat transfer coefficients calculated for a flat unswept wing at zero angle of attack.

In cases where the pressure distribution can be approximated by a power law variation with distance along the surface Bertram[3] has shown how the similarity theory of Li and Nagamatsu can be used to calculate heat transfer rates. This method perhaps rests on a firmer basis than the use of local conditions in the ordinary flat plate formula. In application, too, it is only necessary to know the surface pressure distribution and no explicit calculations have to be made on local conditions at the edge of the boundary layer.

3. Blunt Bodies

If the nose-blunting is sufficiently severe then the inviscid flow field dominates the picture to the exclusion of the boundary layer induced pressure gradient. Then a more direct analytical attack can be made on the laminar heat transfer problem. It has also been found that dissociation effects can be included fairly easily in the analysis. This is because the rate of heat transfer to the surface depends mainly on the difference between the total enthalpy of the air and the enthalpy corresponding to the surface temperature. The Lewis number for air is close to unity and so chemical reaction rates have only a small effect on the heat transfer; for example according to Fay and Riddell the effect of dissociation in the stagnation point case is to increase heat transfer by a factor of $1+0{\cdot}52\ (\mathrm{Le}-1)\,h_D/h_s$ for equilibrium, and $1+0{\cdot}63\ (\mathrm{Le}-1)\,h_D/h_s$ for frozen flow in the boundary layer. Where h_D is the proportion of the stagnation enthalpy outside the boundary layer, h_s, involved in dissociation. With $\mathrm{Le}=1{\cdot}4$ these factors give increases of at most 21 per cent and 25 per cent respectively. One interesting feature however is that if a surface is non-catalytic to atom recombination then the heat transfer rate can be reduced considerably.

In many cases of practical importance the wall is highly cooled, that is, the surface temperature is low compared with that at the edge of the boundary layer. Then the dense layer of air near the surface is little affected by pressure gradients along the surface and a sort of local similarity ensues. This concept has been used by Lees[3] among others. Fig. 3 is plotted from the results of the AVCO team[4], which have been borne out by shock tube experiments, and shows the variation of laminar stagnation point heat transfer with flight speed, nose radius and altitude as represented by ambient density. These results have also been confirmed experimentally by Holder and his co-workers at the N.P.L. but there is some scatter in both sets of tests.

Discarding the approximation of local similarity Kemp, Rose and Detra[5] have treated the problem of laminar heat transfer away from the stagnation region. They conclude that in cases where the pressure distribution does not vary too rapidly then the local similarity assumption as used by Lees gives quite good results.

The presence of a dense surface layer of air has also proved useful in turbulent boundary layer calculations for blunt bodies. The displacement thickness in the case of a highly cooled wall is very small so that the shape parameter H (ratio of displacement and momentum thicknesses) tends to zero. This approximation

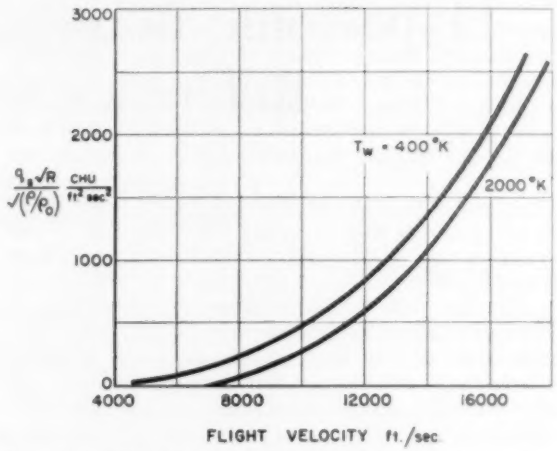


FIGURE 3. Stagnation point heat transfer in laminar equilibrium dissociated flow.

has been employed in several investigations, for example that of Rose, Probstein and Adams[6] who also included the effects of dissociation. A simpler method, accurate enough for design purposes in many cases, is to use the flat plate intermediate enthalpy formula at the true local conditions. This has been shown by Libby and Cresci[7] to be as good or better than many calculations based on more detailed theory, even for fairly strong pressure gradients. It seems likely that dissociation effects too could be included in this method by application of the simple Lewis number factor derived in Ref. 6.

4. Ideal Flat Plate

As has already been indicated, ideal flat plate results often prove useful in pressure gradient cases, at least for heat transfer calculations, provided the true local conditions are used. The intermediate enthalpy method is adequately described in the literature (*e.g.* by Eckert[8] and by Monaghan[9]) and is probably satisfactory for undissociated flow conditions up to about $M=10$, and formulae are available for either laminar or fully turbulent boundary layers. By application of

Laminar

	$\overline{\mathrm{St}_{f.p.}}$	St_c	$\overline{\mathrm{St}_c}$
$\mathrm{St}_{f.p.}$	2	$\sqrt{3}$	$4/\sqrt{3}$
$\mathrm{St}_{f.p.}$	$\mathrm{Re}_x/4$	$\mathrm{Re}_x/3$	$3\mathrm{Re}_x/16$

Turbulent

	$\overline{\mathrm{St}_{f.p.}}$	St_c	$\overline{\mathrm{St}_c}$
$\mathrm{St}_{f.p.}$	1·25	1·15	1·28
$\mathrm{St}_{f.p.}$	0·33 Re_x	0·5 Re_x	0·30 Re_x

FIGURE 4. Relations between heat transfer on cones and flat plates.

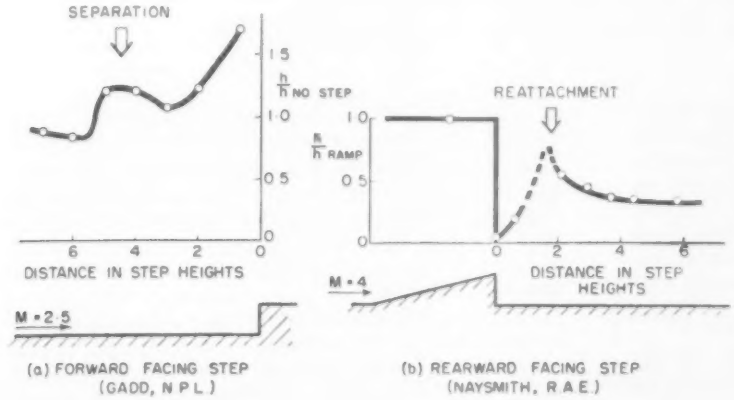
the Mangler transformation in the laminar case, and according to A. D. Young and van Driest for turbulent flow, the flat plate formulae can be used to give heat transfer rates on sharp cones at zero angle of attack. Fig. 4 shows the relations between the mean and local heat transfer coefficients or Stanton numbers on cones and flat plates. An interesting feature of these results is that the local turbulent heat transfer on a cone is only about 15 per cent higher than that on a flat plate for the same local Reynolds number. Since the mean Stanton numbers are integrated quantities we would expect these to be even closer than the local values; in fact the cone value is about 3 per cent higher than the flat plate mean turbulent Stanton number.

To study the effects of dissociation in the laminar case Monaghan[10] has applied Couette flow analysis to the flat plate boundary layer. This yields a simple factor involving the Lewis number and the proportion of energy involved in dissociation remarkably similar to that derived for stagnation point heat transfer in Ref. 4. In both cases the increase in heat transfer is at most about 25 per cent.

5. Separated Flow

Returning to the use of lifting bodies for sustained flight at hypersonic speeds, a typical example is the very narrow delta wing illustrated by Monaghan which may be required to operate at high angles of attack, so that it is now more important than ever to study separated flows. Here we must distinguish between vortex and bubble separation. Also the planned use of bubble separation has been suggested as a means of alleviating aerodynamic heating, for example by a spike protruding from the nose of a blunt body.

Some preliminary research has been done by Gadd[11] and by Naysmith[12] on heat transfer in regions of bubble separation induced by forward- and rearward-facing steps in two-dimensional flow. Fig. 5 shows the variation of heat transfer rates (normalised with respect to the average upstream value) in two typical cases, both with an initially turbulent boundary layer. Although low heat transfer rates are obtained in the separated region behind the rearward facing step, it is particularly noticeable that at reattachment quite high rates were measured; this is also the case at the base of the forward facing step. This result could lead to critical design conditions; it also emphasises that high rates of heat transfer may be expected on the shoulder of a blunt body when a spike is fitted. Thus much of the advantage deriving from the lower heat transfer rates in the separated region may be lost due to increased rates farther aft on the body.



(a) FORWARD FACING STEP (GADD, N P L.)

(b) REARWARD FACING STEP (NAYSMITH, R.A.E.)

FIGURE 5. Heat transfer in separated flow.

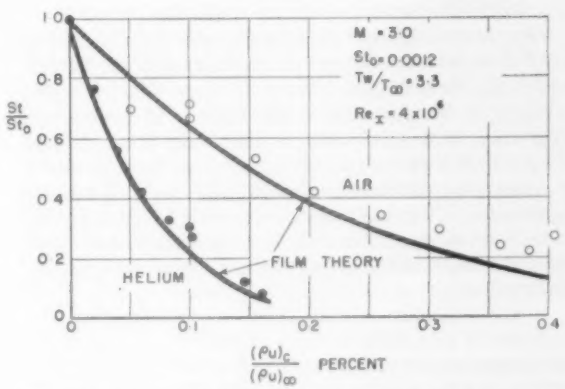


FIGURE 6. Transpiration cooling in a turbulent boundary layer.

6. Transpiration Cooling

Of the many available means of alleviating aerodynamic heating, blunting the nose of a body is perhaps best-known and has been indirectly dealt with above. Other methods include melting or ablating surfaces, film cooling and transpiration cooling. Here we will concentrate on the latter method. Again, simple relations are available, adequate for preliminary design purposes at least; Dorrance and Dore[13] have calculated the effect of air blowing into air for instance. Much current work is in progress, but Fig. 6 is taken from the earlier experiments of Leadon and Scott[14] in a turbulent boundary layer. This shows that a considerable reduction in heat transfer can be achieved with only modest amounts of fluid blown through the wall. At the higher blowing velocities it is even possible to reduce heat transfer below the laminar value. Results with different gases can be correlated to a certain extent by multiplying the mass flow ratio by the ratio of specific heats, coolant to air. However, no really satisfactory theoretical justification of this correlation is yet available.

7. Transition

Our knowledge of the laminar-turbulent transition process is sketchy enough in low speed flow; in the hypersonic flow régime it is even more meagre. The difficulty is of course the large number of parameters involved. As a rough rule of thumb the criterion that laminar flow exists only up to a Reynolds number based on momentum thickness of about 300 has found some application. The condition of the surface however can be such as to make this criterion very inaccurate, even when the effect of other parameters has been correctly predicted.

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Dynamics of a Dissociating Gas

by

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NOTE: *Professor Lighthill's paper was originally given before the First International Congress of the Aeronautical Sciences in Madrid, September 1958. It is reported in full in the published proceedings of that event.* (*Pergamon Press* 1959).
The following condensed version is given to ensure the continuity of this report.

HAVING agreed with Dr. Shercliff that I would discuss dissociation and its applications, after which he would speak about ionisation and its applications, it seemed appropriate that I should repeat the lecture I gave to the International Congress of the Aeronautical Sciences on "Dynamics of a Dissociating Gas"—although with apologies to those few in the audience who may have come back from Madrid with memories of anything except Flamenco and Paella and Goya and Sangría.

The paper began with a detailed study of the dynamics of dissociating oxygen. Two main methods of dissociation exist: one is by collision with other particles, the other is by the absorption of a photon of light, the wavelengths of light required for this latter method being in the range of 1,300 to 1,750 Ångströms. The sun shines powerfully in this range but there is practically no light of these wavelengths present at altitudes below about 60 km. since it has all been absorbed higher up.

At lower altitude dissociation takes place entirely by collision and if the collision is between an oxygen molecule and another molecule many different types of energy may be involved. This large choice of degrees

of freedom increases the possible number of pairs of colliding molecules which satisfy the required total excess energy condition. It is suggested that the number of degrees of freedom involved (n) is 7. Measurements of dissociation rate by Byron[1] indicate that a slightly better agreement is obtained with $n=6$, perhaps because if one oxygen molecule is rotating one way it is only rotation of the other in the opposite sense that can help to give up energy for the benefit of dissociation.

Atoms produced by dissociation may recombine in two ways, the inverse of collisional dissociation being the so-called three-body recombination. The inverse of photo-dissociation is two-body recombination with emission of light. This mode of recombination must predominate at low densities, below 10^{-5} gm./c.c., because triple encounters become so rare, but in regions where aerodynamic forces are important most recombination takes place by triple encounters.

It may be said that the energy required to dissociate one molecule of oxygen is the kinetic energy which the molecule would have if its speed were 5·5 km./sec. Accordingly if a body travels through oxygen at such a speed it can dissociate a large fraction of the molecules that come near to it.

As it passes through the shock wave ahead of the body the kinetic energy of the bulk motion is almost entirely converted to translational and rotational energy of the molecules relative to the mean flow, and dissociation proceeds very rapidly. However, when the equilibrium state is approached, so much of the energy of the molecules and atoms has been absorbed in dissociating collisions, that the temperature has fallen and the dissociation rate is reduced. The changes in rate of dissociation with distance behind the shock wave are so rapid that no direct measurements have a satisfactory accuracy. It is better to assume a formula for this rate of dissociation and determine the constants therein by comparing the inferred density distribution behind the shock wave with that observed by interferometry.

According to these observations, the density increases during dissociation, and this rise in density reduces the thickness of the compressed air cap between shock wave and body. Freeman[2] shows that the stand-off distance of the shock wave depends on the parameter $(\rho_\infty d)/U$, which relates the time taken for a gas to pass by a body of length d with velocity U, to the average collision time. For large enough $(\rho_\infty d)/U$ dissociation becomes complete in a distance small compared with the stand-off distance. For very small $(\rho_\infty d)/U$ little dissociation occurs and there is a thicker cap corresponding to $\gamma=1{\cdot}4$. For such low densities the Reynolds number is small and a viscous boundary layer may well extend right up to the shock wave. One interesting conclusion from recent calculations of this régime[3] was that, if the surface is cooled to a temperature not much above that of the undisturbed stream, then the effect of the boundary layer is to decrease the stand-off distance as the Reynolds number decreases, that is as the boundary layer grows. We may say that the cooled boundary layer has an effectively negative displacement thickness.

At ordinary densities the boundary layer is much thinner and the problem is to establish how the dissociated gas outside the boundary layer affects it. Two points have been stressed in the literature. One is that despite the thinness of the boundary layer, atoms still take quite a long time to cross it and so have time for the necessary triple encounters to enable them to recombine. In fact since atoms cross the boundary layer by diffusion, they do so in about the same time as a portion of gas takes to be convected round the surface of the body. Hence if equilibrium is reached outside the boundary layer it should also be reached inside.

A second well known point is that rates of diffusion of all the different modes of energy are of the same order of magnitude. It follows that, for a given specific enthalpy outside the boundary layer, the rate of heat transfer to the surface will not be altered in order of magnitude by the process of dissociation.

In connection with attempts to calculate the effect more precisely, it may be noted that many estimates use a "Lewis number" of about 1·4 and this is indicated by the theory to be appropriate to a gas which is only slightly dissociated. Lower values may occur in other conditions.

Clarke[4] makes the point that if at lower densities recombination does not produce equilibrium in the boundary layer, then the heat transfer will be reduced. Theories which suggest the contrary have assumed a so-called "fully catalytic" wall where the equilibrium condition is satisfied. However, since there is only a finite rate of collision of atoms with the wall, there is only a finite rate of recombination, and hence a measurable degree of departure from equilibrium is found.

I have selected only a few aspects of the subject for a half-hour lecture and it has been impossible to go into details (*see* Lighthill[5]). The aim could be only to indicate the spirit of the investigations. To sum up, I think their spirit consists of a determination not to look at the problem obstinately as "a fluid dynamicist" or as "a chemical kineticist" but to try to pour out a nicely matured blend of the two.

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Magnetogasdynamics and its possible Aeronautical Applications

by

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1. Introduction—Ionisation in Hypersonics

Professor Lighthill's paper has discussed the dissociative behaviour of air at the high temperatures produced when it is compressed in a shock or heated by friction in a boundary layer under hypersonic flight conditions. Just as molecules may dissociate into atoms, so may these molecules and atoms split further into charged ions and electrons. The equilibrium state of this ionisation process is amenable to the same thermodynamical or statistical-mechanical treatment as in the case of dissociation, the Law of Mass Action in this context being referred to as the Saha equation. Consider the simplest case where a monatomic gas is singly ionised, a fraction α of the atoms having lost one electron, so becoming singly charged positive ions. Then the Saha equation is

$$\frac{\alpha^2}{1-\alpha^2}\left(\frac{p}{p_0}\right)=\left(\frac{T}{T_0}\right)^{5/2} e^{-T_0/T},$$

in which p = pressure, T = temperature and p_0 and T_0 are the characteristic pressure and temperature for ionisation. T_0 is of the order of 150,000°K for common gases, but ionisation occurs well below this temperature because the pressure is usually low. The form of this equation follows purely from the assumption that the atoms, ions and electrons each behave like perfect gases with γ equal to 5/3. The case of ionised, dissociated air is naturally more complicated. Certain oxides of nitrogen which occur ionise particularly easily.

In a hypersonic situation ionisation, like dissociation, may fail to reach equilibrium. In rarefied air the ionisation relaxation process behind a bow shock wave may occupy a region comparable in extent with the vehicle dimensions. There is also the possibility of diffusive energy transport through a boundary layer with ions and electrons recombining on the skin of the vehicle with a consequent heat release to the skin.

Ionisation brings the extra feature, not present with dissociation, that the air becomes electrically conducting. For flight at orbital velocity at high altitude the conductivity of air heated by a bow shock wave is around 100 mho/m., somewhat better than most electrolytes. Typical values of various important quantities are given in Table I. I make no apology for using the M.K.S. system of units, but I should welcome some discussion of the best units to use in magneto-aeronautics. Resler and Sears' interesting paper and its correction[1] give fuller details, and they remark that the conductivity may be significantly augmented by the addition of caesium, potassium or other easily ionised elements, whenever the extra complication is justifiable. Meyer[11] also presents similar data.

When the gas density is low and a sufficiently strong magnetic field is present, a further novel phenomenon occurs. Charged particles in a magnetic field pursue helical paths as they gyrate round the magnetic field lines between collisions. Only if the electron collision frequency is much greater than the electron gyro-frequency does the gas behave like the isotropic fluids with which aerodynamicists are familiar. If electron gyration is significant the transport processes such as heat transfer become non-isotropic, the conductivity perpendicular to the magnetic field being reduced. Similarly electric current flow ceases to be governed by Ohm's Law and the Hall effect occurs. The subject of magnetogasdynamics is generally understood to exclude this type of behaviour, which belongs to plasma physics[2, 3]. However, aeronautical applications will frequently involve such phenomena (*see* Table I), and these will affect the motion and heat transfer to a significant extent.

In magnetogasdynamics as distinct from plasma physics the thermodynamics of the gas is still

TABLE I

SOME TYPICAL VALUES IN HYPERSONICS, ROUNDED OFF TO THE NEAREST POWER OF TEN

Mach number	$M>10$
Altitude	$\simeq 10^5$ ft.
Velocity	$v\simeq 10^4$ m./sec.
Max. flux density in iron	$B\simeq 1$ Wb/m.^{2}
(N.B. electron gyro and collision frequencies are equal when $B\simeq 10^{-2}$ Wb/m.^{2})	
Fluid properties:	
Conductivity	$\sigma\simeq 10^2$ mho/m.
Density	$\rho\simeq 10^{-2}$ kgm./m.^{3}
Viscosity	$\eta\simeq 10^{-4}$ kgm./m. sec.
Permeability	$\mu=4\pi\,.\,10^{-7}\simeq 10^{-6}$
Diffusivity ratio	$\mu\sigma\nu\simeq 10^{-6}$ (dimensionless)
Dimensionless parameters:	
Reynolds number	$R_e=\frac{\rho v l}{\eta}\simeq 10^6$
Magnetic Reynolds number	$R_m=\mu\sigma v l\simeq 1$
Magnetic force coefficient	$C_m=\frac{\sigma B^2 l}{\rho v}\simeq 1$
	(with l taken equal to 1 metre.)
Hartmann number	$H_m=Bl\left(\frac{\sigma}{\eta}\right)^{\frac{1}{2}}\simeq 1$, with l taken equal to 1 mm.

uncomplicated by the magnetic field, because the gas is not magnetic and has the same permeability, μ, as free space. In particular the thermodynamic state of gas in equilibrium may be specified by merely two properties, such as pressure and temperature, and the usual equations of state apply, relating the equilibrium and transport properties.

2. Magnetogasdynamics

When electrically conducting material bears a current in the presence of a non-parallel magnetic field it suffers a body force like that which drives the armature of an electric motor. This then is the main novelty of magnetogasdynamics. We can "grab" the fluid out in midstream instead of merely pushing at its edges as in ordinary aerodynamics. However, the current and magnetic field distributions are not arbitrarily assignable or wholly under our control. The current flow pattern is influenced by the electromotive forces induced by the fluid motion across the magnetic field and moreover any field we impose tends to be modified by the currents in the fluid. Thus the magnetogasdynamic problem essentially concerns the interaction of two fields, the velocity field and the magnetic field. The velocity distribution is affected by the electromagnetic forces while the magnetic field is affected by the currents induced by the motion. There is a superficial analogy with the natural convection problem, with its two-way coupling of a velocity field and a temperature field.

Let us look first at the effect of the motion on the magnetic field. If typical values of the fluid velocity and imposed flux density are v and B respectively, then the induced electromotive forces will be of order vB per unit length. In general, there will also be an electric field E present, associated with a charge distribution (and, in unsteady cases, with induction due to changing magnetic fields). The main function of E is to allow for discrepancies between the distribution of electromotive forces induced by the motion and of the ohmic resistance drops. Sometimes E may be imposed deliberately by electrodes or coils bearing variable currents.

It should be noted that, even if charges are necessary in the fluid to account for E, their other effects are quite negligible in magnetogasdynamics. The electrostatic body forces on the charges and the contribution of charge convection to the current are insignificant. In unsteady cases the Maxwell displacement current and that part of the current necessary to feed the changing charge distribution are also negligible.

In most aeronautical cases, even with E present, the induced electromotive forces vB will produce currents with a density j of order σvB, where σ is the gas conductivity. Such currents circulating in a region of typical length l will produce induced fields H', of order σvBl, and B', of order $\mu\sigma vBl$.

$$\text{Hence} \quad \frac{B'}{B} = \mu\sigma vl = R_m, \text{ say,}$$

a dimensionless number measuring the degree to which the motion distorts an imposed field. It also measures the ability of the motion to induce currents in coils in the vehicle, so as to produce the imposed magnetic field. R_m tends to be rather less than unity in aeronautics. It has come to be called the magnetic Reynolds number because, like R_e, it measures the relative power of the convective and diffusive processes. A magnetic disturbance cannot spread instantaneously into a conducting medium but diffuses into it with diffusivity $1/\mu\sigma = \lambda$, say. Writing $R_m = vl/\lambda$ shows the parallel with $R_e = vl/\nu$ more clearly. When R_m is large, magnetic boundary layers can occur. This is merely the skin effect, well known to electrical engineers, in a new guise. It is fortunate that in aeronautics R_m is not large, since otherwise fields would not penetrate far from the vehicle. The ratio R_m/R_e or $\nu/\lambda = \mu\sigma\nu$ is a dimensionless fluid property, the ratio of two diffusivities like the Prandtl number. In aeronautics it is very small. This means that magnetic disturbances from the vehicle can diffuse much more freely into the air stream than the viscous disturbance, *i.e.* the usual boundary layer. Hence, in studying the viscous boundary layer we may be able to ignore distortion of the imposed field, and in studying the field distortion we can ignore the viscous boundary layer.

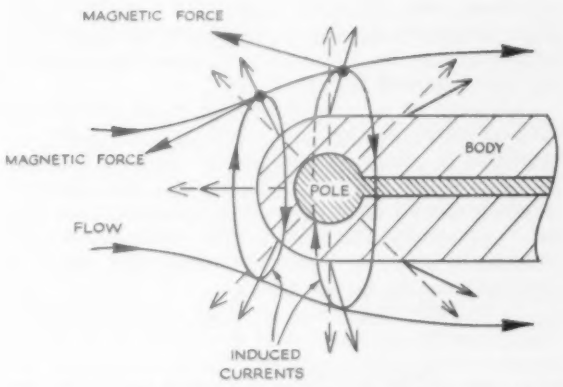

FIGURE 1. Magnetogasdynamic flow past a body with a magnetic pole in the nose. The magnetic field returns to the body farther aft. The imposed field is shown dotted. Its distorted form is shown with solid arrows. In the case shown the electric field is zero and the magnetic force opposes the motion, providing extra drag and energy dissipation remote from the surface of the body. In a hypersonic case the conducting region would not extend ahead of the bow shockwave.

Figure 1 shows a simple aeronautical example of interacting velocity and magnetic fields.

We turn next to the influence of the magnetic force on the motion. The strength of this effect is most simply measured by taking the ratio of a typical electromagnetic force and a typical inertia force, expressed per unit volume. This ratio is

$$\frac{\sigma vB \times B}{\rho v^2/l} = \frac{\sigma B^2 l}{\rho v} = C_m, \text{ say,}$$

a quantity which so far has not acquired a name or regular symbol but certainly deserves them. I suggest the name "magnetic force coefficient" in view of the obvious analogy with other force coefficients in aerodynamics. C_m tends to be rather less than unity in the

hypersonic case, if realistic values of B are used, but is large enough for the flow to be influenced significantly by magnetic forces. These forces oppose the motion in the case illustrated in Fig. 1.

It should be noticed that either of the interactions between the velocity field and the magnetic field may be negligible in certain circumstances. This may appear paradoxical since any force on the fluid must react on whatever device is producing the magnetic field and this implies a distortion of the field. Nevertheless one interaction may be significant while the other is negligible, though not non-existent.

In boundary layers viscous action may dominate, in which case the important ratio is

$$\frac{\text{magnetic force}}{\text{viscous force}} = \frac{\sigma v B^2}{\eta v / l^2} = H_m^2$$

where η = viscosity, l = boundary layer thickness, and $H_m = Bl(\sigma/\eta)^{1/2}$, the Hartmann number, long established in the literature.

In general, though not always, the magnetic force precludes irrotational flow, whenever C_m is not negligibly small. This is one complication. Another is that the motion is never isentropic, even along streamlines, so that the equation p/ρ^γ = constant or its real gas equivalent may not strictly be used. The entropy equation is

$$\rho T \frac{Ds}{Dt} = j^2/\sigma + \text{viscous dissipation} + \text{heat accumulation},$$

in which the last term includes conduction and radiation. Even in the absence of the last two terms, the ohmic dissipation still produces entropy increases.

In steady flow the energy equation is most conveniently expressed in terms of the stagnation enthalpy h_0 for which

$$\rho \frac{Dh_0}{Dt} = \mathbf{E} \cdot \mathbf{j} + \text{viscous work} + \text{heat accumulation}.$$

The term $\mathbf{E} \cdot \mathbf{j}$ represents electrical energy transfer, which causes h_0 to change even in inviscid, adiabatic flow unless E is zero, as in the case of Fig. 1. Thus in this case h_0 is constant outside the viscous and thermal boundary layer. In this case however there is no longer the usual simple relation between vorticity perpendicular to the flow and the entropy gradient, because Crocco's theorem for steady, inviscid motion gains an extra term in magnetogasdynamics, becoming

$$\mathbf{v} \times \boldsymbol{\omega} + \frac{\mathbf{j} \times \mathbf{B}}{\rho} = \text{grad } h_0 - T \text{ grad } s.$$

3. Hypersonic Applications of Magnetogasdynamics

The most obvious application of magnetogasdynamics to hypersonic flight is its use to modify the forces on, or the heat transfer to, the vehicle. Fig. 1 has illustrated a very simple case where the drag is augmented without a commensurate increase in surface heating. The *skin* friction may even be reduced by a magnetic field normal to the surface[4]. A considerable amount of theoretical work has been done on these problems, particularly in America. It is also found that heat transfer at a stagnation point may be slightly reduced by suitably oriented magnetic fields[5, 6, 11]. Heat transfer might also be reduced by exploiting the known tendency of a magnetic field to suppress turbulence. There is the possibility of using magnetic fields to *reduce* drag and achieve high lift by controlling boundary layer transition or separation.

Just as drag may be produced magnetically, so may lift forces and moments. The lifting aerofoil might even be replaced by a coil through which the air could flow virtually unobstructed, suffering only magnetic forces[7]. Alternatively the forces necessary to control the vehicle's attitude might be produced electromagnetically. Another possibility is the use of magnetogasdynamic techniques for instrumentation to indicate air speed, attitude, and so on. Electric control devices and instrumentation should be easy to integrate with a vehicle's general electrical control system.

One interesting case is that where an electric field is deliberately imposed by electrodes so as to supply energy to the air, accelerating it perhaps for propulsive purposes. Alternatively, such a device could run like a dynamo instead of a motor with the air generating electrical energy and supplying it to the vehicle to drive electromagnets, fuel pumps or control actuators (but not to propel the vehicle!). In a propulsive device in which electrical energy is supplied to the air in a duct—an electric ram-jet—more thrust will result if we supply the energy approximately reversibly, propelling the air by subjecting the transverse current to a perpendicular magnetic field, instead of simply dissipating the energy to heat the air. For a given air flow rate and power supply, the smaller the entropy rise of the air, the greater is the thrust. A parameter whose magnitude measures the degree to which the energy can be supplied reversibly at any point is $\sigma v^2 B^2/P$, in which v = air velocity, B = transverse field, and P = power supply per unit volume.

Gas dynamics in the presence of electromagnetic energy exchange presents novel features[8]. For one-dimensional steady flow in a duct of constant cross-sectional area, *supply* of heat or *dissipated* electrical energy drives the Mach number towards unity, and choking results. In contrast, if the energy is supplied reversibly, with the flow, current and magnetic field all mutually perpendicular, the Mach number is driven away from unity. Choking only occurs in this case when energy is *extracted* by dynamo action.

The use of electromagnetic rocket devices seems likely to be confined to interplanetary flight where a low thrust/weight ratio may be acceptable[9, 10].

The weight of electromagnetic equipment is the major problem facing attempts to apply magnetic techniques to aeronautics. Aluminium could replace copper for current-carrying purposes, but the time is ripe for someone to discover a light ferromagnetic material! The weight of electrical equipment is not a hindrance to the use of magnetic techniques for providing high speed or high temperature test facilities on the ground. Electromagnetic driving of shock tubes has already become standard practice. Heat

transfer in hypersonic nozzles might be reduced by magnetic techniques.

Another difficulty is that of achieving a current flow between the air and electrodes on the vehicle, since the air will be cooler near an electrode and will have diminished conductivity. The electrodes must be durable and capable of emitting electrons. One possibility might be to use electrodes in the form of longitudinal fins extending through the cooled air layer. Another possibility is to use inductive coupling of variable currents in the vehicle and the air by transformer action.

4. Conclusion

The outstanding question is this—is magnetogasdynamics likely to be important in hypersonic aviation? One answer is that it will certainly be useful in some applications, providing ground test facilities for example. In more general circumstances, it is difficult to give a positive answer at this stage. In any given situation it should not be too difficult to estimate the order of magnitude of any likely, desired effect and to weigh against any benefit the penalty in terms of weight, complication or expense. It should be clear that magnetogasdynamics is sufficiently interesting and potentially useful to warrant further studies of its application to aeronautics, but it is necessary to stress the necessity of choosing realistic magnitudes. Many of the interesting phenomena which occur in astrophysical or thermonuclear magnetohydrodynamics do not appear in aeronautical situations because the electrical conductivity is not high enough.

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Aerodynamic Problems of Manned Space Vehicles

by

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THE intention of this paper is to point out certain aerodynamic problems that may be of future interest, by trying to argue in general terms about their possible context. Perhaps the most distinctive feature of any project analysis of a vehicle designed to carry a man into an orbit in space and return him safely to the ground, is the need to limit the accelerations and retardations of his passage, and the temperature of his environment. There are of course innumerable other considerations which have to do with his comfort and safety, but it is his inability to withstand excessive heat or high g which distinguishes man from more rugged but less versatile instruments.

There is no need to fear that accelerations in the powered ascent will rise above tolerable limits: assuming a constant rocket thrust and a vertical ascent, Fig. 1 shows that an upper limit to acceleration of $10g$ would still allow an increase in velocity during burning of 1·4 times the effective exhaust velocity, and the possible increase is still higher in an ascent inclined to the horizon, which is of course the practical requirement. Thus with existing rocket fuels (giving exhaust velocities of 9,000 ft./sec.) not even a 2-stage launching rocket would necessarily involve more severe accelerations in achieving orbital speed than the human frame could stand.

After launch we shall assume that the vehicle describes a close orbit about the earth: the problem of deceleration on return from journeys to the Moon or planets will not be discussed here, and certainly the achievement of a close satellite orbit will be the first aim of manned space-flight as it has been with the instrumented vehicles. The radiation belt will provide a powerful argument for the limitation of apogee heights

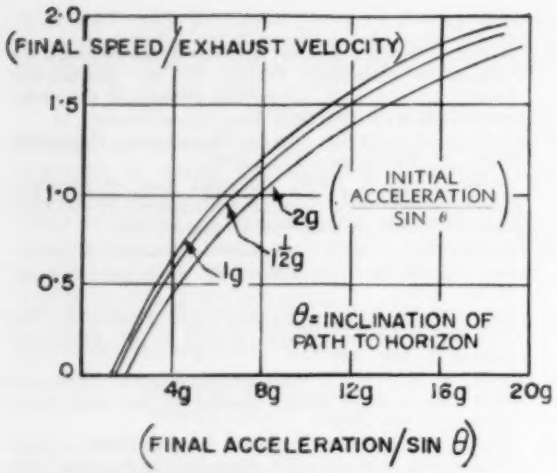

FIGURE 1. The acceleration during powered ascent.

below 600 miles, and consequently the orbit eccentricity is likely to be less than 0·08. As will be shown later, the re-entry path may be precipitated by a voluntary action; but it is convenient for the moment to consider descent from a circular orbit, which is indeed identical with that from an elliptic orbit decayed by the insidious action of air resistance after a large number of circuits have been described.

There are at least two distinct ways by which re-entry may be effected, either by using a glider vehicle (using aerodynamic lift), or by relying on aerodynamic drag alone—the so-called parachute re-entry. Whether or not a "parachute" is used is to some extent immaterial, as provided the vehicle shape and drag coefficient remains constant it can be shown that, in re-entry from a circular orbit, the deceleration always has the same characteristic time variation, reaching a maximum value of 0·8*g*, which is tolerable if uncomfortable for a human. The shape, size and mass of the vehicle only affects the height at which this peak value occurs and not, to any important extent, its intensity (Fig. 2). However, the heat transfer from the boundary layer *is* sensitive to design; pointed or slender shapes are to be avoided as they produce local hot spots, and a low value of (mass/drag area) is valuable in ensuring

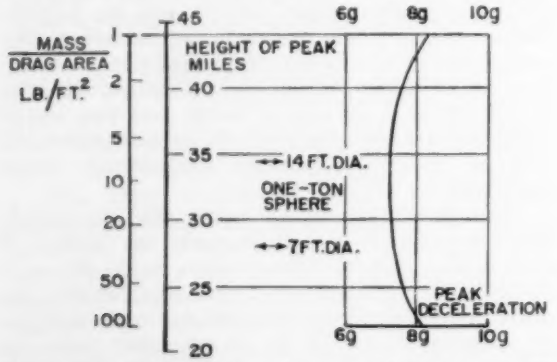

FIGURE 2. Variation of maximum retardation in zero-lift descent as affected by atmospheric properties.

that the maximum heating rate per unit area is low. This maximum occurs at about the same time as the maximum deceleration (Fig. 3), and so with a low ratio of (mass/drag area) it takes place high up where the air is thin. In other words the total heat transmitted depends on the mass, but the amount radiated increases with surface area.

The simplest shape to consider is simply the sphere, imagining this as a pressurised capsule containing the man, and the radiation-equilibrium surface temperature then varies simply as the eighth root of the vehicle density. This temperature would be very nearly constant over the front hemisphere with a mean value 85 per cent of the maximum; Fig. 4 shows the maximum values reached during peak heating estimated from the theory of the laminar boundary layer near a stagnation point.

To place the problem in its perspective, we might envisage a man as needing equipment which, with his own mass, provides a payload of at least 1,000 lb. The diameter of the capsule, which has to be accommodated in the second stage rocket appropriate for this kind of payload, could hardly be more than, say, 14 ft., giving a volume of 1,500 ft.^{3}. The density of the vehicle including its structure would therefore be likely to be about one lb./ft.^{3}, and the maximum temperature of the skin (if insulated internally) would then be close to 1,150°C. This seems just beyond the structural range of present day steels, but a reduction of temperature by even so little as 10 per cent might bring the structural problem into more manageable proportions.

Because the frontal area of the launching rocket increases in broad proportion to the payload and all-up mass, the square-cube law operates and bigger vehicles would suffer a lower temperature. Again with an unmanned vehicle of spherical shape, its auto-rotation in falling would cause the heat input to be spread over the whole surface, thereby reducing the maximum temperature reached by about 20 per cent. But the manned sphere would have to be aerodynamically stabilised (presumably with fins) if only to control the attitude of the man relative to the direction of deceleration. Nevertheless, it would be difficult in practice to prevent the emanation of radiation from the inside

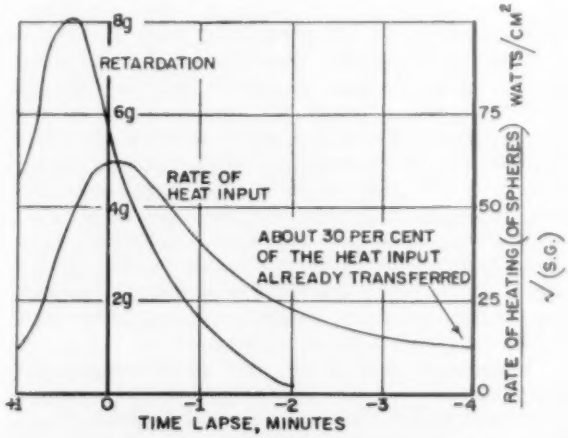

FIGURE 3.

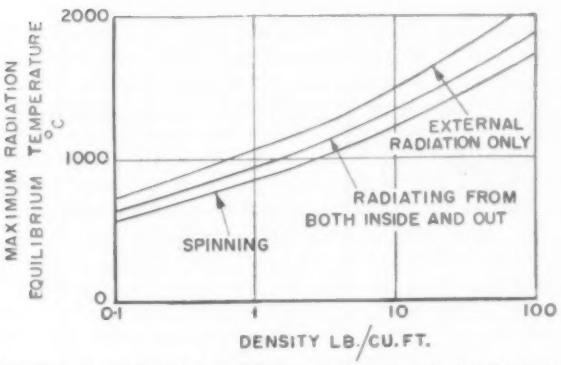

FIGURE 4. Stagnation point temperatures of thin-shelled spheres in peak heating conditions, balanced by black-body radiation.

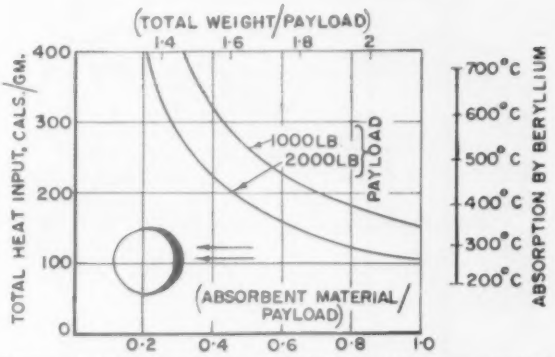

FIGURE 5. Heat input to the absorbing material of a spherical shell of 7 ft. diameter, thickened over its forward hemisphere; and maximum temperatures for beryllium skins. (Structure excluding absorbant material = 15 per cent payload.)

surface of the forward hemisphere, and if indeed this was encouraged to reach black-body proportions, heat could this way profitably escape to the leeward hemisphere, with a reduction of 10 per cent in the absolute equilibrium temperature of the forward hemisphere. The occupant would of course need protection from the radiation and direct body cooling by a refrigerated garment, but this appears a more practical proposition than any attempt to insulate and cool the whole external vehicle shell.

Supposing that its temperature is thus reduced sufficiently to permit tensile stresses in this steel shell of 10 tons/in.^{2} (as a result of internal pressurisation) the structural mass would hardly contribute less than about 0·4 lb./ft.^{3} to the overall density, which on the suggested volume of the 1,000 lb. payload, would imply a total structure of 600 lb.

This figure of course is not intended to do more than suggest the order of magnitude of the structure mass, but it does emphasise that it is likely to be a considerable fraction of the payload, and if the same mass of a material having high heat capacity were used as the shell of a smaller sphere, it would be a sufficient heat sink to limit the temperature below the radiation equilibrium value. In this event, it pays to employ high vehicle density rather than low, as the temperature limit varies in proportion to the ratio of frictional work done to total loss of kinetic energy, that is, to the ratio of skin friction to total drag coefficient. A low altitude retardation is then beneficial as it ensures high Reynolds number and low skin friction coefficient. A spherical vehicle could hardly be less than 7 ft. in diameter to accommodate a recumbent man, and Fig. 5 then shows heat inputs over the forward hemisphere assuming laminar flow in the boundary layer, and 1,000 lb. payload to be carried. For the reasons just mentioned the heat input reduces in proportion to the square root of the mass. If the heat is to be simply stored within, and distributed throughout, the heat sink, a material of reasonably high conductivity as well as heat capacity is needed and beryllium would be appropriate. Maximum temperatures of this metal are indicated in Fig. 5. Even lower values for the same mass could be determined in theory by envisaging other materials as heat sinks leading to the release of latent heat, or the ablation of external insulation, but the practical difficulties of such schemes complicate any realistic estimate of the actual mass required, and the author has nothing new to add about these well-known and promising lines of research.

Absorption limit temperatures are much more sensitive to the accuracy of the heat transfer estimates than radiation equilibrium temperatures, and there is possibly rather greater doubt about the existence of laminar flow at the higher Reynolds number involved in the descent of a high density vehicle; based on free-stream conditions during peak heating and sphere diameter, this equals 1,000 times (lb.wt./ft.diam.), or typically about 10^5; bearing in mind the high rate of heat transfer, and the increase in viscosity of the flow behind the intense shockwave developed by the sphere, one can feel reasonably encouraged to believe that laminar flow should exist. If it does, there is still doubt about the theoretical heat transfer estimates, as quite plainly re-association of the air within the boundary layer exposed to the relatively hot surface must play an important, if questionable, rôle.

Stabilisation of the high density vehicle designed as a heat sink would be rather more difficult, and of course its terminal velocity close to the ground would be high, though even that for the low density radiation shell is fast enough (around 80 ft./sec.) to require landing parachutes. The search for suitable fibres, or the design of metallic foil or mesh parachutes for use throughout the descent seems worthwhile as these would form a very suitable means of stabilisation, and also reduce both the rate and the total heat transfer to the pendent vehicle to a degree which is not limited by considerations of storage space in the launching rocket. The main difficulty would appear to lie in the very high heating rate close to the forward rim of the parachute. Hinged rigid extensions, like the "flared skirt" can be envisaged instead, but would be necessarily more massive.

Variable area devices have been frequently suggested as a means of changing the descent path so that the peak deceleration is alleviated, and this is illustrated in

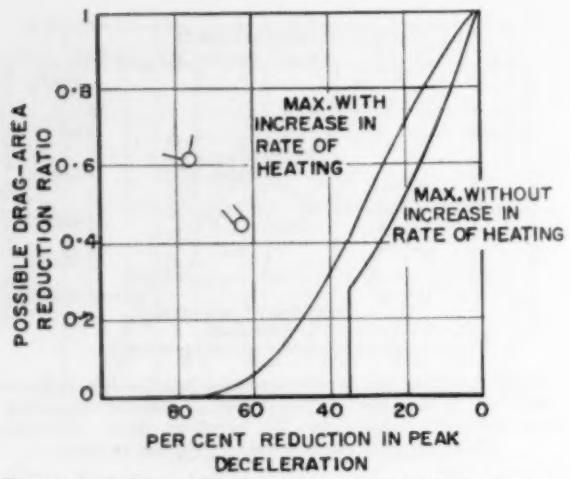

FIGURE 6. Fully-variable drag area, reduced during free-fall, extends the duration, but reduces the intensity of, high g.

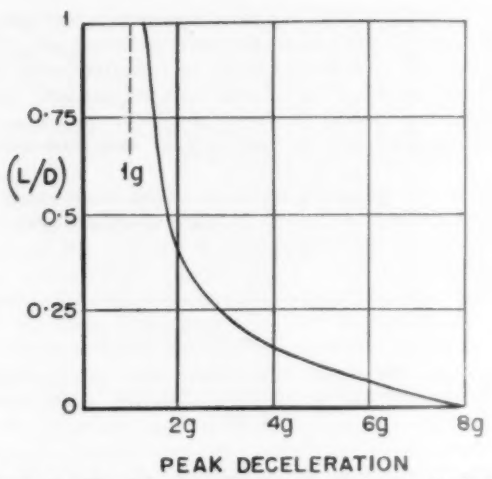

FIGURE 7. The influence of lift/drag ratio on deceleration during re-entry.

Fig. 6. However, it needs a large proportionate reduction in area (suitably timed to take place during the descent) to produce any significant effect, and the temperature of the retained portion of the vehicle is never lower (and usually much higher) than it would be if the large-area configuration were permanent.

The most effective method of alleviating the deceleration is by the introduction of lift (Fig. 7), which also serves to decrease the peak heating rate per unit area, though the total heat absorbed is usually much greater. The equilibrium radiation temperature varies as the eighth root of surface pressure, which is in turn in proportion to wing loading. So wing loading needs to be low, and further it also pays to fly at high lift coefficient, so as to operate in as low an air density as possible (Fig. 8). If these measures are taken the surface temperature becomes relatively low, though it is within 10 per cent of its maximum for some 20 minutes (as opposed to a figure of 1¼ minutes in the zero-lift descent).

The main problem introduced by the heating is that of its unequal distribution because of the sharp leading edge. Inevitably special measures are called for if the edge is not to burn off. The region in danger is admittedly only small—typically within 6 in. or a foot of the nose—but over the longer flight time it receives a sizeable fraction of the total heat input transferred to the whole of the small ballistic heat sink that has been mentioned. The cooling devices which offer promise in absorbing the heat input of these vehicles would be applicable to leading edge cooling, though of course the intense heat gradients in this narrow region are an added difficulty. Indeed if the winged glider is to be competitive with the ballistic vehicle on a weight basis, it would seem necessary to be more subtle and to take advantage of the heat gradient. Direct prevention of the existence of peak heating at the nose may well be possible by means of the forward blowing of a low energy gas stream, and if the effect is needed only over a short surface length, the blowing quantities may not be very great. There have also been suggestions that magneto-aerodynamic effects can be used.

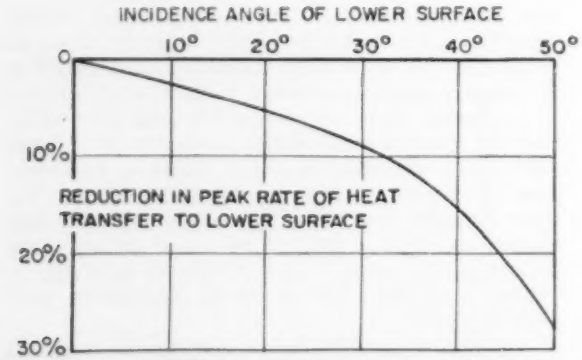

FIGURE 8. Increased altitude of flight results in lower heat transfer rates and higher incidence: a typical example.

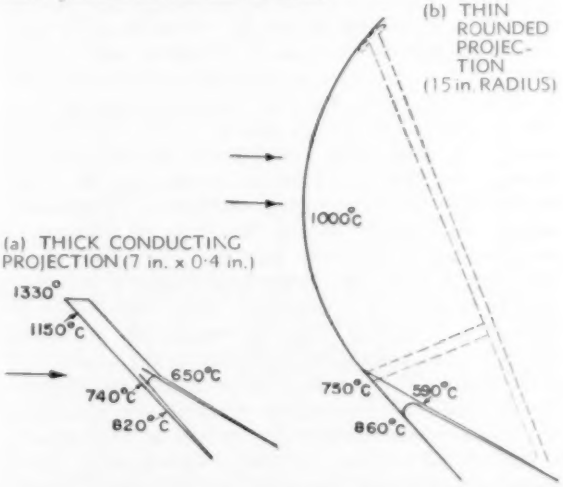

FIGURE 9. Two ways of leading edge heat protection, with temperatures appropriate to wing-loading 10 lb./ft.².

The author has advocated one simple way of taking advantage of the heat gradient by using the conductive properties of a thick metal projection (Fig. 9(*a*)) to transmit the heat downstream and radiate it away from both sides at high but not prohibitive temperature. This device has at least the present advantage that it is amenable to strict theoretical analysis. Mechanical stresses in such thick projections as are needed would be negligible, at least for the lightly loaded glider suggested. Thus a high chromium-molybdenum steel might be allowed to reach 1,150°C, and yet remain serviceable.

However, it may well be that an even simpler solution is to radius the leading edge (Fig. 9(*b*)). Provided we allow the skin material to radiate from both sides, only a 15 in. radius of curvature is needed, for example, to limit the heat input to a wing of 10 lb./ft.^{2} loading so that its temperature is below 1,000°C. However it is difficult to judge how massive this projection should be as it would be determined by buckling characteristics, and would have to be stayed from behind. So for the present argument the conducting projection has been envisaged.

The boundary layer over the nose would be laminar; indeed that over the whole wing surface is likely to be so, as typically the Reynolds number would be about 10^4 per ft. based on conditions behind the intense shock developed in conditions of peak heating. The Mach number outside the layer would also be relatively low, say 4 or 5, and indeed the state of the viscous flow would therefore be in a familiar range, made exceptional only by a high enthalpy ratio across the layer, unusual physical properties of the external flow, and the possibility of re-association of the air close to the surface. It is the last-mentioned consideration which alone casts any doubt on theoretical estimates of heat transfer. Slip-flow effects are not important in such conditions.

The temperature of the flow outside the layer is typically around 5,000°C and one would like to be sure that it is appropriate to ignore radiation from it, as the quantity of air thus heated is considerable. However as atomic gases, and those with symmetric diatomic molecules (including the closely symmetric nitrogen-oxygen associations), have very limited emission spectra in conditions of thermodynamic equilibrium, it is only the presence of water vapour or carbon compounds in the atmosphere, and emission during the dissociation process produced by the shock, which give cause for concern. Unfortunately, as flight under peak input conditions would typically be at a height around 50 miles—in the layer of noctilucent clouds—it would not seem safe to assume that chemical impurities are absent.

At the incidences envisaged—perhaps around 45° or even more—it is inevitable that at high flight Mach number the lower surface would be sustaining the load and receiving virtually all the heat from the air; even if the upper surface were aligned with the free stream, its surface pressure would typically only be 0·2 per cent of that on the lower in peak heating conditions, and the rate of heat transfer 5 per cent. In practice, a much larger quantity of heat would be radiated to this surface

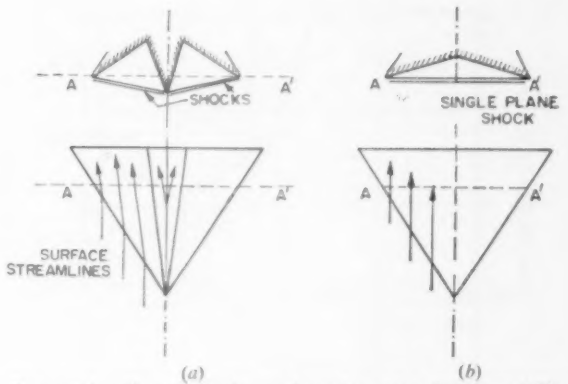

FIGURE 10. Shapes of wing undersurface of a delta amenable to treatment by exact shock-wave theory, with two (or one) plane shocks (schematic only).

from below, and as in the instance of the radiating sphere, this escape of heat from the hot side is to be encouraged.

The high wing-loading would encourage an all-wing design and something resembling a delta planform, involving a relatively small aerodynamic centre shift over the extensive range of flight Mach number, might well be favoured; (with certain reservations, the heating problem is unaffected by the shape of planform used). In assessing the effects of dissociation on the flow of course the exact inviscid gas-dynamic theory must be used, and the conical flow over a delta is a considerable simplification in this endeavour. Various discrete models of exact flows over the underside, based on the postulation of plane shocks from the leading edge, can be easily constructed (as shown, for example, in Fig. 10) and may serve as a useful basis for a perturbation: in practice one would want to avoid the lower meridian becoming a line of shock attachment as it would then suffer the same heating problem as the leading edge; this is avoided and constant pressure preserved over the under surface by making it recessed (as in Fig. 10(*b*)), giving the effect of anhedral. The anhedral on the upper surface would be even more marked (Fig. 11) but the author has found no simple examples of the exact expansive flow over this side of the delta wing: presumably at high incidence a cavity lobe will

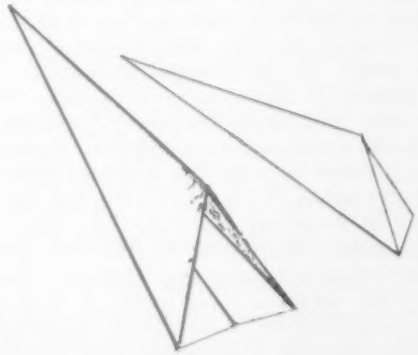

FIGURE 11.

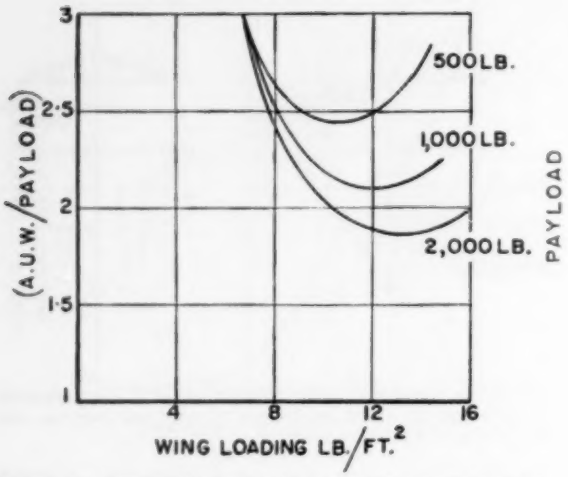

FIGURE 12. Some estimates of the weight of a re-entry glider.

extend from the leading edges, but its shape is not easy to establish, as there is an inflow, and a recompression, towards the upper meridian.

The flat-based wedge section enables the aerodynamic load, and with it the aerodynamic heating and surface radiation, to be spread over all the underside: the wedge thickness can be large without disturbing the high speed properties owing to the near vacuum over its back, and this great depth would be a structural advantage. The pressurised capsule containing the occupant could evidently be submerged within the wing if necessary.

Quite plainly the very unusual shape may well have unusual flying qualities at low speeds! At hypersonic speeds all aircraft suffer a more or less complete absence of aerodynamic damping; thus free motions will be harmonic oscillations or divergences and true stability will inevitably be the responsibility of the pilot (or auto-pilot). Thus for a delta with small positive $-m_w$ and n_r, the disturbed motion at high incidence will consist of rapid incidence adjustment and Dutch roll oscillations of about 10 and 3 seconds period respectively, together with a virtually neutral degree of freedom in angle of roll. Such oscillations would not be difficult to control, though Dutch roll may become too rapid for comfort if $-l_r$ or n_r are made too large. Response in incidence, yaw and roll could easily be made of normal rapidity. With stick free, extra oscillatory modes (if not divergences) exist in all these three degrees of freedom, and aerodynamic balancing would need to be carefully arranged to separate the periods and avoid resonant divergence. In the quoted design, it should be noted that a high fin might well be completely useless, but from what has just been stated, a small fin projecting below the wing would be all that is needed, though it would need heat protection.

Some tentative weight assessments have been made of the glider suggested, fixing its span at 12 ft., its root trailing edge thickness as 10 ft., and assuming various payloads—consisting of the man-carrying capsule. The results are shown in Fig. 12. The structure is envisaged

Wing:			
Skin	220 lb.		
Frame	550 lb.		
Joints	40 lb.		
Projection	120 lb.		
		Total Wing	930 lb. (43 per cent)
Controls:			
Elevons	80 lb.		
Fin	40 lb.		
Circuits	20 lb.		
		Total Controls	140 lb. (6 per cent)
		Undercarriage	110 lb. (5 per cent)
		Capsule	1000 lb. (46 per cent)
		TOTAL	2180 lb.

FIGURE 13. Typical weight breakdown of re-entry glider.

as of steel with a stressed skin supported by a spanwise and chordwise lattice of triangular ribs: in all conditions examined the strength of such a wing, with its short span and great depth, is much more than adequate for the design flight conditions, involving an equivalent air speed of only about 80 ft./sec.; the weight is determined solely by stiffness considerations, provided the nose temperature is limited (below 1,150°C) by the conducting projection described earlier. At low wing loadings the structure weight becomes of course relatively excessive, while at high loadings the projection becomes unduly massive. The optimum appears somewhere about 10 lb./ft.²

Again we see that at best the structure mass is likely to be about equal to the payload carried, and so even heavier than that of the ballistic vehicles described. A sample breakdown of the weight is shown in Fig. 13, and in this illustration the projection consists of a strip 7½ in. long by 0·4 in. thick. In fact a 20 per cent reduction in projection weight would only change the maximum temperature reached by one per cent or 14°C, so its

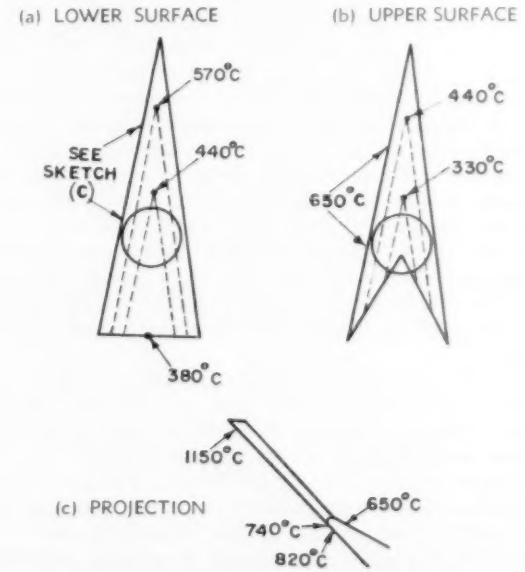

FIGURE 14. Surface temperature distribution.

precise weight is relatively unimportant: however, by the same token, it would be difficult to limit the temperature further.

It should be mentioned that the assessment of heat transfer close to the leading edge is not easy, due to shock-wave interaction, but pessimistic assumptions have been made and a 10 per cent error in estimation of heat transfer makes a difference of 3 per cent in maximum temperature. The temperature distribution in the condition of peak heat input (at a speed of 20,000 ft./sec.) is indicated in Fig. 14, and presuming that the manned capsule would be near the centre of area, it will be seen how successful this re-entry method is in limiting the temperature of the occupant's environment (to about 500°C average.)

All that has been said so far assumes that re-entry is made from an orbit: however it is an awkward but inescapable fact that far more stringent conditions are likely to be met in a premature re-entry due to failure of the launching rocket, particularly if this occurs during the second-stage firing. Thus in the matter of the peak deceleration of a ballistic descent, which depends equally on rate of descent as on speed, the steeper descent into the altitude region where drag is high of a slower re-entry can cause severe retardation. For instance, values of 50 and 60g have been suggested for the descent of I.C.B.M.s, as opposed to the 8g value in descent from a circular orbit. The heating rates of the ballistic trajectory may be likewise increased, but not the total heat transmitted, which is of course a point in favour of the high density absorbing shell as opposed to the radiating shell. However, if a man is to be carried, obviously a way must be found of averting this contingency, as premature re-entry would be (at least on present standards) a distinct hazard and he cannot withstand very high g. Similar embarrassments would be presented to the glider pilot if he found himself diving even at a small angle towards his aerodynamic ceiling: he could pull out and follow a skip trajectory, but even if the gs of this manoeuvre disturb neither him nor the structure, and if the skip is short enough that the structure does not heat up as a whole, nevertheless the leading edge will undoubtedly do so, to its detriment.

The solution would appear to be to lower the all-burnt altitude as far as possible, and so also the perigee of the first orbit (if successfully attained). This can only be done by omitting the period of free ascent between the firing of the two stages, (which is usually introduced to attain the desired final perigee height) and by utilising as high a thrust/weight ratio as the human occupant can accept. A typical ascent path with these measures introduced is shown in Fig. 15: lower altitudes than the one given (60 miles) could be attained by providing lift force from rocket deflection, which is not assumed however in working this example. Fig. 16 shows the peak decelerations suffered in a ballistic re-entry from various positions of the second-stage burning programme, in which it will be seen that the low altitude deceleration of the high density sphere is a great disadvantage which might well rule it out of consideration. The principles of absorption cooling can of course be applied to a larger sphere, but the material

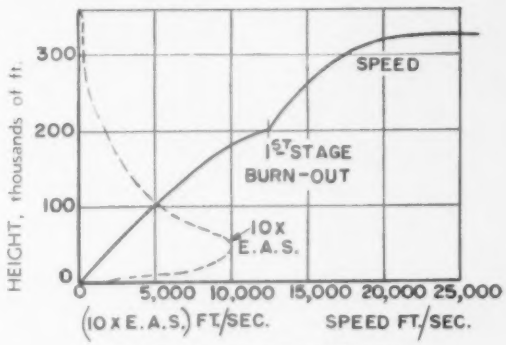

FIGURE 15. Low ascent trajectory.

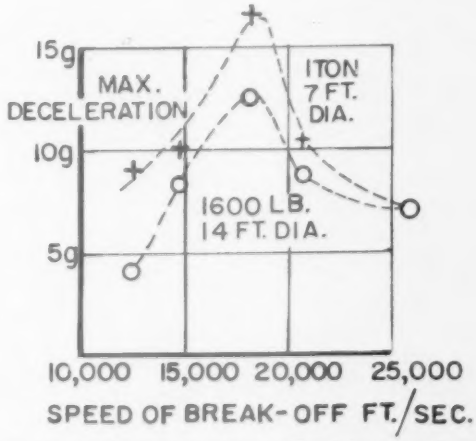

FIGURE 16. Maximum retardation suffered by two spheres, in premature re-entry from launch path of Fig. 15.

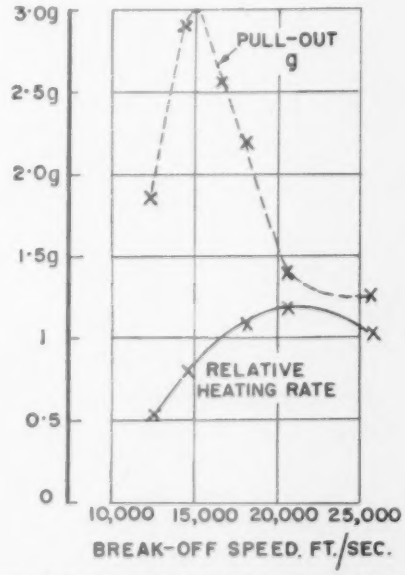

FIGURE 17. Pull-out g during the first dive from an abortive descent path assuming constant C_L and $(W/C_LS)=15$ lb./ft². Also shown is the increase in peak rate of heat transfer relative to that in descent from orbit. (Launch path of Fig. 15.)

weight would have to increase in rough proportion to the radius, to preserve the same temperature.

Figure 17 shows the pull-out g developed on a glider trimmed to a constant lift coefficient, and the increase in its nose temperature compared with the values in re-entry from an orbit. Quite plainly the embarrassment of premature re-entry is severe even in this low launching trajectory.

Such a low perigee altitude as this (*i.e.* 60 miles) might be inappropriate for an orbit: if so, a small third stage rocket could be fired at the apogee of the first circuit; an increase of speed of a mere 60 ft./sec. is then sufficient typically to increase perigee height by 40 miles, and such a small impulse would only require an expenditure of one per cent of the re-entry vehicle's mass in rocket propellant. In any event, the firing of a retro-rocket of similar impulse at the last perigee would be necessary to precipitate descent, if the natural process of orbital decay is too long to await. Other small rocket jets would be needed for stabilisation in space, whose overall propellant mass is likely to be of the order of a few lb./hour of orbital duration.

The precipitation of descent by retardation at apogee is certainly effective with the glider configuration, as it can assume flight under negative lift at the reduced perigee level at super-circling speeds, while drag effects further retardation. Provided this perigee altitude is chosen with due care, there is no need for the heating problems to be more severe than those encountered at sub-circling speeds (at least with the apogee heights below 600 miles). It may be necessary to invert the plane to have the "under" surface exposed to the heat as at sub-circling speeds.

On the other hand, no such control may be exercised upon the ballistic vehicle which appears to suffer appreciably higher retardations (and rates of heating) in this kind of precipitate descent. The evidence is by no means conclusive that this is necessarily so, as it depends upon many factors, including the precise impulse provided at apogee; but in a descent from a 600 mile apogee, peak retardations of up to $12g$ have been worked out. Bearing in mind the undesirable effect of such decelerations on the occupant, it may be necessary instead to reduce the speed by discharging retro-rockets close to perigee, and so reduce apogee altitude as a means of, or before, precipitating re-entry. This of course, would avoid the trouble, but it would also result in a large penalty in the propellant needed —perhaps up to 10 per cent of the vehicle weight. However, before this is admitted, the problem needs careful investigation.

If it were the only solution, it would largely destroy the one single advantage (of lighter weight) which the radiating ballistic vehicle has over the glider. The absorbing ballistic vehicle, with its temperature adjusted to give comparable heat input to the location of the occupant, and with its diameter made large enough to eliminate unduly high g in emergency re-entry, might be heavier than a glider carrying the same 1,000 lb. payload—at least if nothing better than a metallic absorbant is developed. Of course the weights suggested have been hazarded solely for the sake of argument rather than final judgment: many people see virtues in the glider despite its weight, though these frequently imply a refined, fully aerobatic aircraft. On the contrary it is debatable whether it need even be capable of landing; after all an airfield would be a remarkably elusive target in an unpowered glide which, as a result of emergency or mistake, might be initiated over the Pacific Ocean. Possibly it would be better to eject the capsule when low down to land by parachute in sea, mountain, or jungle. Fundamentally the glider itself should be regarded as only a refined form of parachute, particularly suited for high speed operation, but only a rudimentary aeroplane.

Summary of the Discussion

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During the discussion the point was made by a speaker from the Industry that he had come to the meeting to learn; that many of the audience of 200 agreed with him was reflected in the discussion, for, of the 23 who participated 19 were from the research establishments and universities.

EXPERIMENTAL FACILITIES AND MEASURING TECHNIQUES

In reply to several questions concerning the accelerometer balance shown in his film, Dr. Holder explained that the aerodynamic load was applied to the model suddenly, as an impulse, and from measurements of the acceleration of the model and sting the force on the model was deduced. Since the balance was as yet in the early stages of its development, little data was available on its performance but the intention was to compare the drag of a disc measured by the accelerometer balance with that derived from pressure measurements. It was hoped that the starting time, of the order of 10 microseconds, would have only a small effect on the balance readings.

In discussing the merits of free flight rocket testing a member of the audience observed that the use of solid fuels had meant that over the past two years considerable progress had been made in exploring the lower fringes of the hypersonic speed range. One problem was the lack of control over an experiment, but if this difficulty could be overcome by suitable planning, free flight testing could be extremely useful as it enabled the results of shock tube and shock tunnel testing to be extrapolated. Another speaker enquired whether the "short-duration" test facilities ever in fact had constant properties. He also asked whether present or proposed facilities were satisfactory for testing bodies with ablation or with transpiration cooling. Dr. Holder agreed that much research was required to study

flow uniformity; the N.P.L. had already spent 18 months on this problem. For research on ablation and allied problems it was necessary that running times should be increased beyond those provided, for example, by shock tunnels. The high enthalpy tunnel using storage heaters or the arc-heated tunnel might provide solutions. To reproduce real gas effects for long running times it would probably be necessary to use an arc-heated jet, a plasma jet or a device where the walls were protected by magneto-gasdynamic effects. Useful results had been achieved in the study of ablation by the use of models made from materials with low melting points tested in tunnels working at modest temperatures. Concerning transpiration cooling, it seemed profitable to attempt to understand the fundamentals of the problem with the hope of avoiding the expense of a new test facility.

CHARACTERISTICS OF HYPERSONIC FLOW FIELDS

A questioner from Imperial College enquired how it was possible to divide a slender body into four régimes for the purpose of obtaining its pressure distribution, it would seem that this was very difficult for a body which differed from those on which pressure measurements were available.

Dr. Cox replied that the regions indicated on Fig. 8 applied only to bodies similar to the one shown. The blast wave analogy was strictly applicable only to a particular family of bodies having profile of the form $y \propto x^n$, however, the method could give an engineering approximation to the way in which pressure varied over the body for fairly slender aft body shapes.

A question was asked concerning the degree of applicability of the Newtonian flow concept to flat-faced bodies with sharp edges; the questioner thought that perhaps Dr. Mangler's Fig. 2 exaggerated the lack of agreement between the experiment and the "Newtonian" solution. Dr. Mangler was inclined to agree with this last remark, but thought that although a skilful use of the Newtonian solution allowed the experimental values to be aproached, there would always be a considerable limitation on its use for really blunt bodies. A member of the audience suggested that if the shock shape were not greatly different from the shape of the body nose then the Newtonian solution agreed well with experiment, although of course the shock itself did not enter into the Newtonian concept. A further point was made that the shock shape was intimately connected with the value of γ in the gas; Dr. Cox had shown the effect of a change of γ from 1·67 to 1·4, but in a dissociated gas γ could be as low as 1·1 or 1·2. Since the governing parameter was the difference between γ and unity, was the value of $\gamma = 1{\cdot}4$ in fact representative of the gas condition in the gun tunnel? Dr. Cox replied that it was; with a stagnation temperature of 1100°K for the shock wave shape test, only the vibrational modes would be excited and the relaxation times involved would be short enough to ensure that the flow in the nozzle was in equilibrium throughout its length.

One of the restrictions on Dr. Mangler's work was that it proceeded from a given shock shape to a body shape and he was asked whether it was possible to calculate the shock shape associated with a given body shape, *e.g.* a blunted cone. He replied that so far a set of solutions had been obtained for conic sections so that shock shapes for given body shapes could be found by interpolation within this family. The major difficulties arose on those parts of the body which were in the transonic and subsonic region. For shapes such as blunted cones much of the body would be in a region of supersonic flow and a method existed whereby it was hoped to obtain families of solutions for specific bodies.

A speaker from the R.A.E. enquired into the relationship between the work of Cox and Mangler using continuum concepts, and that of Freeman described by Professor Lighthill. Would this latter, which he understood to be on Newtonian lines but taking account of dissociation, supersede the former? In reply Professor Lighthill explained that Freeman's work was on the lines of the Newton-Busemann (or Newtonian + centrifugal) theory but was modified so that consideration was given to a gas particle passing through a shock wave and moving along one of the stream lines; instead of using equilibrium thermodynamics for the change of stability of the particle, non-equilibrium equations (including equations for the rate of dissociation) were used. The "Newtonian + centrifugal" concept was known not to be as accurate as Dr. Mangler's work. The effects of dissociation were fairly considerable and could not be allowed for by merely taking a corrected value of γ since the gas was not in a state of equilibrium.

A question was asked as to whether areas of large vorticity in the flow were always close to the wake. Dr. Mangler replied that these areas arose mostly in association with strongly curved shocks. Farther away from the body where the shock was less curved, the vorticity or loss of total head was much reduced, so that the shear layer would be confined to regions close to the wake.

HYPERSONIC BOUNDARY LAYERS

The point was raised that the results of Dr. Crabtree's first reference had been obtained at a Mach number of approximately 4 and the speaker questioned the advisability of using these results to predict behaviour at hypersonic speeds; Bogdonoff's work, even forgetting entropy effects, indicated that hypersonic conditions really only appeared at Mach numbers greater than 10. Dr. Crabtree was of the opinion that the results could be extended as far as Mach numbers of 10, but it would be dangerous to try to extrapolate them beyond this.

Dr. Crabtree was asked for his views on the apparent tendency for heat transfer rates to be high in a region of reattachment. He replied that the indications were that when the flow reattached, *e.g.* behind a rearward facing step as in Fig. 5 (*b*) of his paper, a small region of circulatory flow appeared, causing the flow to be re-energised within the bubble. This flow gave up its energy in the form of heat to the surface and as it circulated was again energised from the external flow. At reattachment the effect was almost as if a hot jet were striking the surface. Behind the reattachment point conditions gradually returned to those of normal turbulent heat transfer to a flat plate; it was difficult to predict conditions for shapes other than flat plates.

Considerable discussion followed a request for comments on the flow past a complete wedge or cone with no supporting surface behind it, *i.e.* no sting or splitter plate. The suggestion was made that Naysmith's results could be regarded as though for the upper half of a double wedge with a reflection plane along the centre-line. Thus a similar bubble would be formed in the wake of the lower wedge, resulting in almost a second stagnation point at the base on the centre-line. This could be a critical design condition if the heating rates at this point approached the values at a blunted nose. Another speaker thought that the flow behind the wedge would in fact be a series of vortices corresponding to the low speed, low Reynolds number phenomenon of a Kármán vortex street.

Experimental evidence would be required to resolve this, but measurements of heating rates did tend to support the former suggestion. Another speaker thought that at supersonic speeds there would always be a contracting wake giving a trapped bubble.

A third speaker mentioned Ackeret's work which had found temperatures higher than stagnation behind semi-cylindrical bodies at low supersonic speeds. It seemed likely that this was due to unsteadiness of the flow. This speaker enquired whether any information was available on concave noses; he offered the suggestion that a local eddy might in effect insulate the surface and reduce the very high heat transfer rates in the region of the nose. At very low speeds Ackeret's experiments had shown a temperature less than free-stream at the base of the body.

In connection with the problem of separation and re-attachment, a question was asked concerning the change from vortex- to bubble-type separation. Was anything known about the mechanism of the change? Did it occur at a different incidence as Mach number increased? Dr. Crabtree replied that very little was known about vortex type separation at hypersonic speeds, but it was assumed to exist since all surfaces were behind the bow shock. It would probably be possible to answer the question on variation of incidence with Mach number in the near future when current experiments were concluded.

The maintenance of a laminar boundary layer was a powerful way of reducing heat transfer, another method was to try to induce separation; it was interesting to speculate on the effect on heat transfer in the separated region of the state of the boundary layer upstream of separation, *i.e.* whether laminar or turbulent. Dr. Crabtree's opinion was that it would make very little difference.

Concerning the effect of transpiration cooling on transition, it appeared that the coolant would induce transition earlier than it would otherwise occur.

DYNAMICS OF A DISSOCIATING GAS

In replying to a comment on the negative displacement thickness mentioned in his paper, Professor Lighthill said that although results had been published on stagnation point boundary layers with negative displacement thicknesses, the interesting feature of Herring's results was that the actual displacement thickness as calculated by the ordinary formula was positive, but as far as the effect of the boundary layer on the flow was concerned it was as if the displacement thickness were negative. This was because ordinary boundary layer theory was only approximate for the displacement-thickness effect of thick boundary layers. The effect of increasing boundary layer thickness was to bring the shock nearer to the body in low density conditions ($R_e \simeq 5{,}000$).

MAGNETOGASDYNAMICS

Dr. Shercliff was asked how it would be possible to pick up a current from the flow past a vehicle. He replied that the ionised air could be passed through a duct and if a magnetic field existed across the duct it would be possible to induce an e.m.f. The problem would be to obtain satisfactory contact between the walls and the gas stream. Considerable time had been spent on this problem in the United States but no satisfactory solution had been obtained. It might be possible to have longitudinal fins protruding through the thermal boundary layer in the duct but their life would be short.

Alternatively, transformer action could be used to couple currents in the gas with those inside the vehicle but this would require the magnetic field to be oscillatory.

In ordinary rocket or jet engine exhausts the degree of ionisation would be too low, unless augmented by the addition of caesium or a similar substance, to give any significant body forces.

The magnetic drag coefficient of about unity given in the paper corresponded to conditions at an altitude greater than 100,000 ft. and Mach numbers of the order of 20. C_m was not much affected by increasing Mach number, but if the density became too low the region of plasma physics was approached and the gyratory behaviour of the particles became a significant problem, *e.g.* the conductivity of the gas was reduced transverse to the magnetic field.

A speaker from the R.A.E. suggested that at very high altitude the combined effects of aerodynamic and magnetic drag would be small, although perhaps it would be possible to seed the air to increase its conductivity. Dr. Shercliff was inclined to agree with this comment and suggested that this lent support to the argument that it was not obvious that magnetic techniques had a worthwhile application in flight. His own opinion was that they could best be used in high enthalpy test facilities.

AERODYNAMIC PROBLEMS OF MANNED SPACE VEHICLES

In discussing the procedure for descent from a satellite orbit Mr. Nonweiler enlarged upon the problem of speed control during descent. Assuming a circular, *i.e.* decayed elliptical orbit using retro-rockets at apogee, a 60 ft./sec. reduction in speed would reduce the height of the perigee by 40 miles.

A speaker from the R.A.E. thought that within the time of descent (20 minutes) from an altitude of 60 miles the heating rates were not high enough to achieve radiation equilibrium when conduction into the structure was allowed. It would probably be necessary to use some sort of insulation to keep the payload temperature down to 60°C. He also questioned whether in fact there would be a reduction of heating rates by decelerating at lower altitudes, as suggested by Mr. Nonweiler.

For aerodynamic reasons he would have preferred a shape such as the frustum of a cone to a sphere and he would have thought that on entering the atmosphere there would be an enormous increase in longitudinal stability due to the exponential build-up of density. Much could be said in favour of a non-metallic material for the vehicle, *e.g.* durestos; heat would be absorbed in removing the surface and the substance would also act as its own insulator.

Mr. Nonweiler agreed that true radiation equilibrium temperature would not be reached, the values given in his paper were rather high. One would need insulation about the pilot's capsule but in the analysis he had envisaged internal radiation as an aid in reducing skin temperature. If the skin temperature were allowed to rise then by all means use insulation and dispose of the radiation externally: this would make it easier to keep the occupant cool.

On the problem of reduced heating rates at lower altitudes, heating depended primarily on work done against friction. The ratio of the work done against friction to the total loss in kinetic energy (which was fixed) was equivalent to the ratio of skin friction coefficient to drag coefficient, and to make this ratio small it would pay to reduce the skin friction coefficient. If no transition to turbulence were assumed, this meant it would pay to increase Reynolds number and decrease altitude.

On the question of stability of ballistic vehicles he had been considering auto-rotation due to eddy shedding, and not stability in the aircraft sense.

Aircraft in Agriculture*

An Assessment of Their Impact on New Zealand Farming

by

R. H. SCOTT, B.Com.

(Land Utilisation Officer, Department of Agriculture, New Zealand)

OF THE 43¼ million acres occupied for farming in New Zealand approximately 28½ million acres or 66 per cent are too steep to be cultivated by implements. The productive capacity of much of this country can be improved by oversowing with clovers and grass seed, and the application of superphosphate, but there was little opportunity to do this on an extensive scale other than spreading laboriously by hand until aircraft were used for the purpose.

It was fortunate for New Zealand hill country farmers that aircraft came to their aid after the Second World War. With the period of full employment that has obtained since that time, casual labourers in rural areas have been attracted to work in cities, and it has become practically impossible to get men to undertake work such as hand topdressing of hill country. Without some mechanical aid to do this work, considerable areas of hill country would have deteriorated and there would have been an inevitable decline in production from these areas.

The rapid adoption of aircraft as an aid to New Zealand farming after the war has been due to five main factors:—

1. The hilly and steep nature of much of the farming land.
2. The inability to get labour to undertake hand topdressing and seeding of hill country.
3. The much improved economic position of hill country farmers as a result of the higher prices ruling for wool and meat during, and after, the war.
4. The number of skilled pilots who were available.
5. The small training aircraft which were then surplus to requirements and which could be bought at a reasonable cost and adapted cheaply to carry fertiliser.

History of Agricultural Aviation

Although agricultural aviation is a recent development, it is not easy to say when the first attempt was made in the Dominion to use an aeroplane as a farming implement. One of the earliest attempts was probably in 1936 when a sheep farmer owning a coastal hill farm in Hawke's Bay decided to save himself the task of spreading seed by hand on a steep hillside where there had been some slipping after heavy rains. Two canvas bags of seed fitted with spouts were lashed to the aeroplane and the seed released over the area to be seeded.

In 1938, a Mr. A. Pritchard who was employed as a pilot by the Public Works Department made experiments sowing lupin seed on sand dune country to the north of Auckland, and in May of the next year a test at fertiliser dropping was made. The war prevented any further tests being made and it was not until 1946 that the first aerial topdressing experiment took place in the Dominion. At Thames in that year copper sulphate was successfully spread from the air. In September 1947, the Royal New Zealand Air Force entered the scene and an aerial topdressing test was made at Ohakea Air Force Station in the North Island, using a Grumman Avenger aircraft which was adapted to carry 26 cwt. of superphosphate. This trial was in a very light wind and results worked out by the Department of Agriculture showed that a mean density of 2·7 cwt. per acre of granular superphosphate was achieved over an area of 54 acres. Crosswise measurements indicated that, although spacing between flights was not perfect, it was sufficiently accurate to ensure that relatively small areas received not less than one cwt. per acre, or more than five cwt. per acre, and almost all the fertiliser was well distributed within the target area.

There were subsequent trials by R.N.Z.A.F. machines which showed that aerial topdressing was operationally possible and the way was open for commercial exploitation. In 1949, the first private commercial concern made tests. Progress has since been rapid, as shown by Table I.

In addition to aerial topdressing, for the year ending 31st March 1958, aircraft spread 17,260 cwt. of pasture seed over 163,470 acres, dropped 4,300 tons of rabbit poison and distributed 723,500 gallons of insecticides, fungicides and weed killing material.

The part that aircraft play in topdressing the Dominion's pastures is shown by the fact that of 890,000 tons of phosphatic fertiliser manufactured by New Zealand fertiliser works for the year ending 31st March 1958, nearly 48 per cent was spread from the air. For every 10 acres of grassland topdressed, the aeroplane has been doing approximately 4½ acres.

The capital investment in the agricultural aviation industry is considerable. There are approximately 75

*A lecture given before the Society on 5th May 1959. Because Mr. Scott was unable to be in England at the time, the lecture was read, on his behalf, by Mr. C. V. Dayus, Agricultural Adviser, New Zealand Agricultural Branch, London.

TABLE I

NEW ZEALAND AERIAL TOPDRESSING

Year ended 31st March	*Weight of fertiliser and lime distributed* (tons)	*Area treated* (acres)
1950	5,003	48,741
1951	44,957	428,737
1952	88,869	802,212
1953	144,802	1,376,118
1954	203,110	1,929,499
1955	279,006	2,783,802
1956	404,933	3,853,169
1957	428,245	3,945,586
1958	465,733	4,159,464

operators with 280 aircraft and, with ancilliary equipment like fertiliser loaders, their capital investment is about N.Z. £1·3 million. In addition there is the investment made by farmers in constructing airstrips on their properties. An official figure for 1957 showed that there were 5,716 farm airstrips in New Zealand and recent estimates give a figure closer to 8,000. From cost data compiled by the Department of Agriculture it appears that the average construction cost of an airstrip is £350 and, allowing for fertiliser sheds erected near the strips, the average total cost is probably about £400. Assuming there are now 6,500 farm air strips, the aggregate investment by farmers is likely to be about N.Z. £2·5 million.

Uses of Aircraft as an Aid to Farming

AERIAL TOPDRESSING

This is the major use of aircraft in the farming field in the Dominion and, of about 75,000 hours flown on agricultural work in the year 1957-58, 65,800 hours or 88 per cent was occupied on aerial topdressing.

Phosphatic fertiliser is usually applied at about 2 cwt. per acre; a recent survey to obtain data on aerial topdressing costs indicated that about 54 per cent of the farmers in the Dominion put on this quantity per acre. However, there were some variations; in North Auckland most of those having aerial topdressing done applied 3 cwt. superphosphate per acre, while in Canterbury the application rate was generally one cwt. per acre. Although substantial quantities are distributed by aeroplane in each month of the year, the graph shows a distinct peak in the autumn months.

In the early stages of aerial topdressing nearly all the fertiliser was delivered to farms in bags and stored on the farm airstrip. On some of these strips, platforms were erected level with a lorry tray to reduce handling and the fertiliser was stored on this platform under tarpaulin covers.

The position is now altering, and a considerable quantity of fertiliser spread by aircraft is taken from the fertiliser works to the airstrip in bulk. However, much of it passes through bulk fertiliser stores sited at strategic points alongside railways. The fertiliser is taken in bulk in railway wagons from the fertiliser works to these stores and then carted in bulk by motor lorries,

Flat and easy rolling ○ Hilly and mountainous ●

FIGURE 1.

as required, to farm airstrips. Some farmers have erected small sheds or bins on their airstrip to hold the loose fertiliser and the aerial topdressing operators have the necessary mechanical loading equipment to handle the materials. In the North Island there are 65 bulk fertiliser stores with a capacity of nearly 28,000 tons. In the South Island so far there is only one such store in operation.

Operating charges for topdressing vary from district to district, but within districts charges are fairly uniform. Typical aircraft charge rates per hour for aerial topdressing are: —

DH-82	£12-£14 per hour
FU-24	£26 per hour
DH Beaver	£34 per hour

However, the farmer is more interested in the all-in cost on a per acre basis and to obtain data on costs

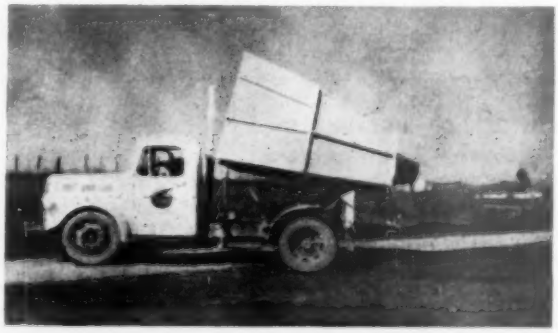

FIGURE 2. A dump truck which weighs the load of fertiliser on a hydraulic cell. (Accuracy ± ½ per cent.)

TABLE II

AVERAGE COST OF AERIAL SPREADING OF SUPERPHOSPHATE AT 2 CWT. PER ACRE

Cost Components	*Waikato Bay of Plenty King Country*	*Gisborne Hawkes Bay Wairarapa*	*Taranaki Rangitikei Wellington*	*Nelson*	*Canterbury*	*Otago-Southland*
	£ s. d.	£ s. d.	£ s. d.	£ s. d.	£ s. d.	£ s. d.
Fertiliser ex works	19 10	1 0 11	1 0 10	1 0 4	1 0 4	19 10
Railage on fertiliser	3 11	2 7	3 9	5 5	7	5 1
Cartage to airstrip	3 4	3 6	1 1	5 11	3 4	2 4
Aerial spreading by aircraft	6 6	6 0	6 0	7 7	7 7	7 1
Interest on capital investment in airstrip	7	9	1 3	1 6	7	3
Other*	3	6	9	2 4	4	3
Total cost	1 14 5	1 14 3	1 13 8	2 3 1	1 12 9	1 14 10

*Includes such costs as annual maintenance of airstrip and cost of labour.

to the farmer, the Department of Agriculture carried out a survey in 1957. Information was collected from a number of farmers in hill areas where aerial top-dressing is a feature. The average costs by districts are set out in Table II.

Some adjustments were made to these figures to make them reasonably comparable, *e.g.* the cost of fertiliser was that as for superphosphate in bags. The costs can be considered as approximating those which most farmers in their respective districts had to pay. There are, of course, extremes and a recent study made by C. P. Tebb, Field Officer at Hamilton, for the New Zealand Meat and Wool Board Economic Service, showed that in Waikato costs on a per acre basis to spread 2 cwt. per acre of bagged superphosphate ranged from an all-in cost of £1 6s. 11d. per acre to £1 16s. 0d. per acre. In the former case the road haul was short, whereas in the latter case the haul was longer and the conditions for air work were not favourable.

Although over the past five years there have been increases in most of the cost components—fertiliser, railage and road cartage charges—there has been little difference in the per acre cost to the farmer. This has been due to such factors as the mechanical handling and loading of fertiliser, the increased efficiency of pilots and aircraft, and the greater number of farm airstrips that are now available. This latter factor has decreased the flying time between airstrips and target areas. An indication of this is shown by the quantity of fertiliser spread for each hour flown by aerial top-dressing aircraft. For the year ended 31st March 1951, 2·79 tons of fertiliser were distributed during each hour flown and for the year ended 31st March 1958, the average amount distributed per hour had reached 7·07 tons.

AERIAL SEEDING

Two methods of aerial seeding are practised, distribution of seed mixed with fertiliser and pure sowings. The first method, which is usually restricted to clover seeding of existing pastures, is to mix the seed with the fertiliser in the hopper of the aircraft. The second method is to distribute seed as a separate operation and, for this, special attachments are fitted to the hopper. From an amount of 48 tons of seed distributed over 18,700 acres in 1951, the amount reached 8,130 tons over 163,470 acres in 1958. A small part of this is done on new blocks of land being developed by the two State departments, the Lands and Survey Department and the Maori Affairs Department. The former department has approximately 8,000 acres per annum of newly developed country seeded by aircraft, which is about one-sixth of the total area developed by this department each year.

A typical cost for aerial seeding is 5d. per lb., which means that a seeding rate of 20 lb. per acre would cost 8s. 4d. per acre.

AERIAL FENCING AND SUPPLY DROPPING

The spreading of fertiliser and seed on hill country is the initial stage in the development of this country, but much other work then remains before its ultimate productive capacity can be attained. The principal subsequent requirement is the closer subdivision of farms, for without this it is impossible to manage the

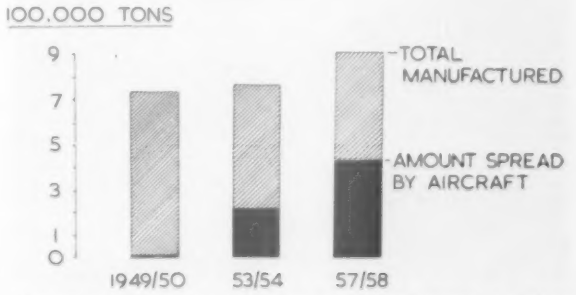

FIGURE 3. Phosphatic fertilisers manufactured and spread by aircraft.

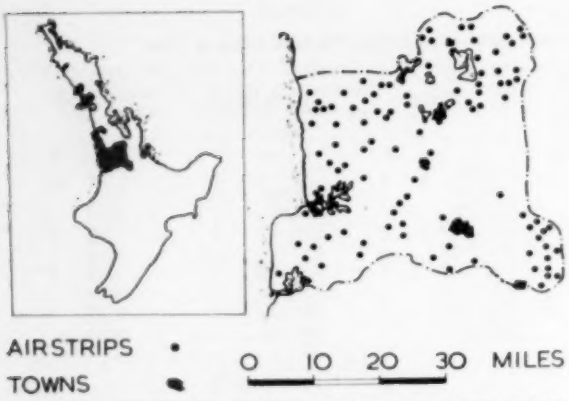

FIGURE 4. Distribution of farm airstrips in a North Island district.

improved pastures effectively. To carry out the closer subdivision of much hill country in the past the farmer was faced with the laborious and slow task of carting fencing materials to fence lines on pack horses, but during the past three years aircraft operators have entered the field of supply dropping and for the 1957/58 year 8,100 tons of fencing materials were dropped from the air and 210 tons of other supplies. If economic conditions for farmers are favourable in the future, this work is likely to increase considerably.

SPRAYING

The application by aircraft of sprays to control insects, plant diseases and weeds, has made considerable progress over the past four years, the most important aspect being aerial spraying to control weeds. Of 723,500 gallons distributed from the air for the year ending 31st March 1958, some 608,200 gallons or 83 per cent of the total were used for weed control purposes.

Because of the toxic qualities of many of the materials used, aerial spraying presents special problems and to assist aerial operations the New Zealand Agricultural Chemical Manufacturers' Federation in 1958 prepared a guide for operators in the handling and application of chemicals.

To supplement the spraying work being done by orthodox aircraft, there are now six helicopters in operation in the Dominion.

A recent development whereby aircraft may be of considerable assistance in developing some of the hill country is in the practice which has become commonly known, but misnamed, as "chemical ploughing." It is a method whereby low fertility grasses like danthonia and browntop, which are common on New Zealand hill country, can be destroyed by the application of about 5 lb. of the chemical Dalapon and one lb. Amitrol per acre. A pasture seed mixture with the necessary fertiliser can be distributed on the litter. This work can be done by aircraft and it appears that, whereas it costs £18 to £20 per acre to establish a pasture on hill country by cultivation methods, by using an aircraft to kill the existing low-fertility sward and spread the seed and fertiliser, the total cost is likely to be about £12 per acre. Much experimental work has already been done in testing this procedure, and if it proves sound in practice, it should allow the economic development of large areas of the Dominion's surface sown hill country.

Costs to develop this class of country must be as low as possible, as the potential carrying capacity of much of it is about 3 ewes per acre against potential capacities of 6 to 7 ewes per acre on many flat areas. On the other hand, to develop hill areas by the orthodox method of cultivation is greater than the cost of developing most flat areas by somewhat similar methods.

RABBIT DESTRUCTION

Rabbits were an early problem in New Zealand, for they became a serious pest in the early 1870's and they remained so until recent years. It was not until 1947 that a determined effort was made to cope with the problem, legislation being passed in that year under which a Rabbit Destruction Council was set up as an advisory body to the Minister of Agriculture. One of the Council's main functions is to devise and promote measures to destroy and control the pest. Up to that time the trapping of rabbits for their skins was often a profitable undertaking, but because of the levy imposed on the rabbit skins sold in New Zealand, and the

FIGURE 5. A Cessna 180 aircraft topdressing typical New Zealand hill country.

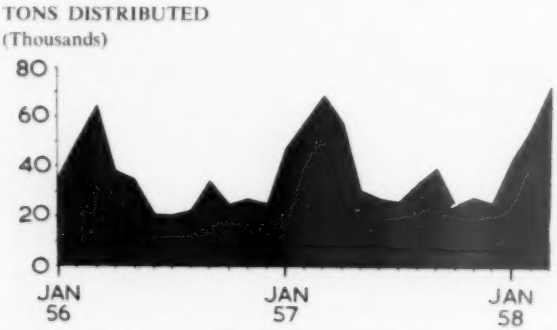

FIGURE 6. Aerial topdressing between January 1956 and March 1958.

prohibition of the export of rabbit carcasses and their sale locally, the "farming" of rabbits has no longer any attraction and the policy now is to destroy by poisoning. The "killer" policy has been most effective.

Aircraft have assisted in the work of poisoning the pest, particularly in areas which have difficult access. One example of rabbit destruction work by aircraft is that being done on a 120,000 acre mountainous sheep station in the South Island. Carrots are grown on the station as the bait with which the poison is mixed. To enable the necessary cultivation and harvesting of the 60 acre crop to be done, a tractor and digger were flown into the station and three aircraft are employed during the winter to spread the poisonous bait. At the present time the station carries 10,500 sheep during the winter, but when the rabbits have been eliminated it is estimated that the station will carry approximately 30,000 sheep.

Over the past three years, aircraft have distributed an average of 4,500 tons per annum of poison bait to aid in the destruction of the rabbit. A more recent development has been the dropping of bait in areas where opossums are a serious pest.

Physical Results

In a dynamic agricultural economy, a number of factors contribute to the increase in productivity of farms. In New Zealand it is not possible, on the basis of existing farm management data and the present analytical techniques, to isolate one factor and determine its contribution to any productive increase. Aerial topdressing and seeding of hill country are no exception. These are only two of the factors contributing to the increase in the productivity of New Zealand's hill country. Admittedly they are frequently the initial contributing factors, but others contribute later before the optimum productive capacity is attained, such as additional subdivisional fencing and the installation of more stock water facilities.

One can look at the picture in a general way by tracing the increases which have occurred in the number of sheep and beef cattle since 1950, when aerial topdressing and seeding began on a commercial scale (*see* Table III). Dairy cows are excluded as dairying is a feature on flat and easy rolling country where little aerial topdressing and seeding are done.

FIGURE 7. Spraying blackberry and lupins.

TABLE III

SHEEP AND BEEF CATTLE NUMBERS IN NEW ZEALAND

Year	*Sheep*	*Beef Cattle*
1950	33,857,000	2,088,000
1951	34,786,000	2,149,000
1952	35,384,000	2,282,000
1953	36,193,000	2,478,000
1954	38,011,000	2,634,000
1955	39,117,000	2,808,000
1956	40,255,000	NA
1957	42,382,000	2,861,000
1958	(1) 46,000,000	NA

(1)—Provisional

NA—Not available

During this same period the output of phosphatic fertilisers increased from 703,000 tons to 972,000 tons per annum and of the total output of fertilisers during the nine-year period, aircraft spread 2,060,000 tons or 21 per cent.

It has been assessed that about three-fifths of the total increase of 12 million sheep and about 800,000 head of beef cattle came from hill country, and as most of the fertiliser spread by aircraft was on this class of country, it might appear that the aeroplane was the major contributing factor to the stock increases achieved. However, other factors played a part, such as the near eradication of the rabbit, and the use of heavy crawler tractors and implements to cultivate hill areas, which could not previously be cultivated with the equipment available. Blades fitted to the front of crawler tractors have also done much in constructing tracks on hill country to give access to areas where it was previously impossible to take cultivating equipment. That aircraft have contributed to the stock increases is certain, but the extent of the contribution is impossible to determine. However, there is one aspect which is sometimes overlooked; if aircraft had not entered the aerial topdressing field it is fairly certain that the productive capacity of some of the Dominion's hill country would have declined, as there was not the labour available for hand topdressing and oversowing of the country. It would have reverted to undesirable vegetation such as manuka, bracken fern and gorse.

To obtain factual material on the impact of aircraft on the Dominion's agriculture one can look at case studies or at the results of surveys, and here again this material suffers from the problem that the results are clouded by other issues, and by improvement work on the farms which has taken place following aerial topdressing and oversowing. However, two case histories will serve as an introduction, these being two farms where the evidence is strong that aircraft have played a major role in the results achieved up to the present time.

The first concerns a property of 3,300 acres in North Canterbury, ranging in altitude from 300 ft. to 1,500 ft. For a period of 35 years until 1950, the carrying

TABLE IV

STOCK INCREASES AND WOOL PRODUCTION ON NORTH CANTERBURY SHEEP FARM

Year	*Ewes*	*Hoggets*	*Cattle*
1950	1,150	230	30
1957	1,650	330	200

capacity of the property was practically static. In that year a start was made to topdress and oversow a block of 575 acres from the air. Subterranean clover, cocksfoot and ryegrass were spread, as well as 15 tons of superphosphate. Further blocks were topdressed and oversown until a total of 2,520 acres had been treated by aircraft.

The increases in stock numbers which have occurred as a result of this work are shown in Table IV.

The increase in wool production was about 4,800 lb. per annum, which, if it realised an average of 30 pence per lb., meant an increase of about £600 per annum from this source alone.

The second case history is that of a property of 1,740 acres in Marlborough, which ranges in altitude from 700 ft. to 3,400 ft. There are about 190 acres of flat, but rather broken, country and the balance is fairly steep hill country.

When purchased by the present owner in 1942, the property was carrying 1,170 sheep. In the 1945-46 season a start was made to develop the land on the flat portion of the farm, and by 1950 sheep numbers had been increased to 1,500. During that season a start was made at aerial topdressing, and 160 acres were topdressed with superphosphate, this being the start of the hill country improvement work. From then until the end of the 1956-57 season some 750 tons of superphosphate, 12,800 lb. clover seed (subterranean, white and red clover) and 10,200 lb. cocksfoot seed had been spread by aircraft. Most of the property had then been topdressed, some of it more than once. The results from the 1949-50 season, when the aerial work began, until June 1957 are given in Table V.

Wool production has doubled in seven years, and from this source alone the increased revenue is approximately £1,800 to £2,000 per annum. Cattle were first run on the property in 1954-55 and now there are over 50 head.

Although there have been these considerable increases, the owner of the property states that the limit is not yet in sight.

TABLE V

SHEEP SHORN AND WOOL PRODUCTION ON A MARLBOROUGH SHEEP FARM

Season	*Number of Sheep Shorn*	*Wool Production* (lb.)
1949-50	1,500	14,140
1950-51	1,568	13,995
1951-52	1,490	17,200
1952-53	1,717	19,900
1953-54	2,037	19,460
1954-55	2,183	21,700
1955-56	2,211	25,720
1956-57	2,445	29,890

Aerial Topdressing Surveys

The most comprehensive survey so far carried out in New Zealand to assess the physical and economic results from aerial topdressing and oversowing was one done by Mr. C. P. Tebb, Field Officer of the New Zealand Meat and Wool Board Economic Service at Hamilton. This investigation covered a period of five years from 1952-53 to 1956-57 and was conducted in a hill country area in the Raglan, Waikato and King Country districts of the North Island. Of a total of 56 farms in the survey, 21 were selected as "fully aerial," that is, there were few factors other than aerial topdressing and oversowing which contributed to the results achieved.

Some of the basic features were as follows: —

(*i*) The total productive acreage of the twenty-one properties was 16,165 acres in 1952-53 and 16,291 acres in 1956-57, the increase of 126 acres being due mainly to some formerly unproductive land being brought into production on six farms.

(*ii*) Of the total productive acreage of the farms, 3,085 acres or approximately 20 per cent could be cultivated by implements. In effect, most of the productive acreage could be improved only by hand methods or by the use of aircraft.

(*iii*) The average size of the farms was 769 productive acres, the sizes varying from 306 productive acres to 2,400 productive acres.

(*iv*) In the 1952-53 season when the five-year analysis began the topdressing status of the farms was—

Two farms had topdressed aerially for one year previously.

Two farms had topdressed aerially for two years previously.

One farm had topdressed aerially for three years previously.

(*v*) The area of crops grown on the farms over the period was negligible.

PHYSICAL RESULTS

The physical results were assessed on the basis of the wool produced (Table VI) and the number of stock wintered.

This represents a 17·7 per cent increase over the period and on the basis of wool production per productive acre the increase was from 18·7 lb. to 21·9 lb.

The position regarding stock wintered is set out in Table VII. To simplify the comparison over the years,

TABLE VI

WOOL PRODUCTION ON SURVEY FARMS

Season	*Wool Production* (lb.)
1952-53	303,834
1953-54	315,585
1954-55	340,061
1955-56	328,185
1956-57	357,763

TABLE VII

STOCK WINTERED ON SURVEY FARMS

Year	*Total Ewe Equivalent*
1952-53	45,551
1953-54	47,396
1954-55	49,422
1955-56	49,576
1956-57	50,666

all stock on the farms were converted to a common unit, the "ewe equivalent." This is done on the basis of the feed intake of the animals; for instance it is usually said that one beef breeding cow is equal to 5·5 ewes.

The increase of 5,115 ewe equivalents over the period was equal to an 11 per cent increase and, if looked at from a productive acre point of view, it represents an increase from 2·82 ewe equivalents per acre in 1952-53 to 3·11 ewe equivalents per acre in 1956-57. This is something in excess of one quarter ewe per acre.

FINANCIAL RESULTS

While the financial returns from the sale of sheep, wool and cattle on a per acre basis showed an increase from N.Z. £6 1s. 6d. per acre in 1952-53 to N.Z. £8 7s. 3d. per acre in 1956-57, this made no allowance for changes in market values. To make the figures comparable the market influence was removed and the 1956-57 values were equated back to 1952-53. This showed that the bare extra cash return from the increased volume of production in 1956-57 as compared with 1952-53 was 12s. per acre.

On the farms in the survey the application of superphosphate per acre averaged 2½ cwt. per acre, and the all-in cost to have this applied (*i.e.* cost of fertiliser, freight on fertiliser to farm airstrip, and aerial spreading charges) was £1 18s. 9d. per acre.

As an average of 66 per cent of the acreage of the farms was topdressed during a year, this meant that by spreading the topdressing costs over the whole farm acreage, the cost was approximately £1 5s. 6d. per acre. On this basis it meant that the 12s. extra cash return per acre was paying approximately half of the topdressing costs.

One of the conclusions of the survey reads "Topdressing at any level is not normally undertaken unless it is covered by the farming income, along with all other farming expenses; therefore the fact that half of the topdressing cost is being immediately recouped after a certain level is built up must surely be highly satisfactory."

FIGURE 8. Take-off from an airstrip at 2,300 ft.

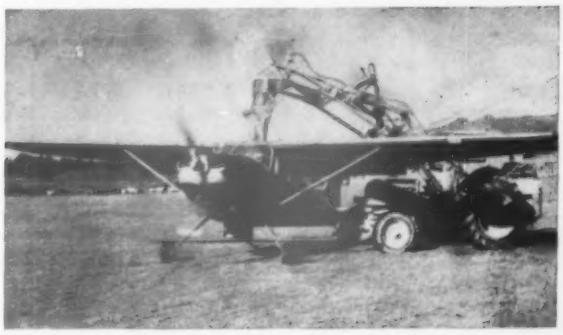

FIGURE 9. Loading a Cessna 180.

This conclusion may be correct, but it is yet too soon to be certain. Other work such as subdivisional fencing may still be required before the optimum capacities of the properties are reached. For all this work to be sound economically, the additional net return from the extra livestock carried must more than meet interest on capital expended to carry out the improvement work to attain these optimum capacities. Some aerial topdressing work can come into the category of capital expenditure as the initial applications of fertiliser raise the fertility of the soils, and thus enables better pastures to be established.

Information which the Department of Agriculture has obtained from two small groups of farmers, who have been carrying out aerial topdressing, shows how difficult it is to assess the contribution of aircraft to the increases in production which have occurred. For seven farms in the Waikato and King Country district, areas of which ranged from 640 acres to 3,150 acres, the average carrying capacity per acre increased from the equivalent of 1·74 ewes in 1954-55 to the equivalent of 2·10 ewes in 1956-57, and the wool production per acre from 10·7 lb. to 11·8 lb. For the other small group of four farms on the East Coast of the North Island, which

FIGURE 10. Loading an Auster Agricola.

ranged in size from 320 acres to 2,060 acres, the average carrying capacity increased from the equivalent of 2·04 ewes to 2·19 ewes, and the wool production per acre from 14·3 lb. to 14·8 lb. However, on all these properties some improvement work had been done before aerial topdressing began in 1954-55, and such work will continue, as well as other types of development work.

An indication of the follow-up work involved where hill country farmers have started improvement programmes, was obtained from another survey made by the Department of Agriculture of some farms selected at random on the East and West Coasts of the North Island. All of the farmers have done some form of improvement work during the past 10 years. Information was collected from them in 1958. Of 51 farmers visited only one considered that he had completed his improvement programme, the remainder still having a considerable amount of improvement work to be done. Thirty-three farmers, or about 65 per cent, considered that the erection of additional subdivisional fencing was now the major work to be completed. Of course, in some cases, the fencing material might be dropped on the fence lines by aircraft.

Problems

The main problem now facing the New Zealand hill country farmer is that with the lower prices which have been ruling for sheep, lambs and wool over the past two years, he may not be able to continue with his development and improvement programme and maintain his topdressing programme. This can have a serious effect on agricultural aircraft operators, and at the time of writing this paper (February 1959) there has been a considerable decline in the amount of work being done by operators.

Another problem which some hill country farmers must face is the necessity to bring up to a better standard their farm airstrips. Investigations made by the Department of Agriculture two years ago indicated that many of these strips were in an unsatisfactory condition. In a survey of 132 strips selected at random in the Auckland district many had some defect; for example, 69 or 52 per cent had a defect in construction, such as being graded unevenly, and 53 or 40 per cent had surface defects, such as gullying and erosion, which had developed after construction. In addition, some had been sown down with unsatisfactory seed mixtures; in fact, only 10 per cent could be classed as being satisfactory in this respect. These were only some of the results of the survey.

Usually farmers get aircraft operators to decide on a satisfactory site for a strip, but usually no other guidance has been sought by the farmer when having a strip constructed, and in any event technical guidance has not always been available.

Conclusion

The evidence so far available on the results of aerial topdressing and oversowing indicates that it is yet too soon to evaluate its real impact on New Zealand agriculture. Until a considerable number of hill country farms attain more stable livestock carrying capacities nearer their potentials, it will not be possible to assess the economics of the improvement work carried out, and even then the part played by aircraft will be only one of the factors contributing to the results achieved.

Nevertheless, aircraft have assisted in the improvement of large areas of hill country and certainly without them the productive capacitity of some farms on this country would have declined, instead of showing a considerable improvement over the past decade.

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APPENDIX

THE AIRCRAFT IN AGRICULTURE

(By courtesy of Civil Aviation Administration, New Zealand)

There is no doubt that the ability of aerial topdressing to establish itself and expand so rapidly resulted from the availability in 1949 of a very large number of R.N.Z.A.F. surplus DH 82 aircraft. There was little demand for this training equipment—the Aero Club requirement had been satisfied—and purchasers could almost name their own price. This position, in relation to aerial agriculture, was not peculiar to New Zealand as a similar situation obtained in the U.S.A. in the immediate post war years when American operators benefited substantially from thousands of Stearman PT 225 h.p. aircraft which were thrown on the market at very low figures.

In the period 1949-1952 the Tiger Moth with conventional hopper installed in the front cockpit was virtually the only type of aircraft used in aerial work, but the payload limitations imposed by the normal Certificate of Airworthiness resulted in serious economic problems for the operators; at a relatively early stage there was an insistent demand for the application of US CAR 8 procedures, with their permissive overload and relaxation from many stringent requirements governing orthodox operations.

The New Zealand authorities, after careful examination of CAR 8 and CAM 8, were of the opinion that these United States procedures would not, under local conditions, ensure the margins of safety considered desirable. At the same time, however, it was conceded that aerial topdressing could not be an economic proposition unless payloads were

increased. The problem was given some study and finally a formula was evolved for an overload condition, this being carefully calculated in respect of each aircraft type on the relationship between structural strength of the airframe and the actual performance of the aircraft. This practice enabled the all-up weight of, for example, the DH 82 to be increased by nearly 300 lb.

Notwithstanding this important concession operators appreciated that the DH 82 could be considered an interim aircraft only. The nature of the terrain being flown, the standard of airstrips available and the demands of the job called for an aircraft of better performance, more rugged construction and greater payload. From joint discussions with Government, operators evolved a specification for the "ideal" aeroplane. The principal characteristics included: —

1. Hopper payload not less than 1,000 lb.
2. Gross weight 3,000-4,000 lb.
3. Take-off distance 1,000 ft. at 3,000 ft. altitude.
4. Initial rate of climb—1,000 ft./min. at sea level.
5. Low stalling speed.
6. High degree of pilot safety; special strengthening of cockpit area.
7. All parts to receive anti-corrosion treatment.
8. Rugged construction—particularly in regard to landing gear.
9. Seating for additional two or three persons (loaders, spare pilots, etc.).

A general survey of the field in 1952 disclosed no aircraft likely to meet the requirements of operators and, following representations from the operators, the Minister in Charge of Civil Aviation set up a special committee comprising representatives of the Aviation Industry Association, the New Zealand Federated Farmers and the Civil Aviation Administration, to consider the various aircraft available or projected and from these select a type as a replacement for the DH 82.

The Committee conducted a survey of aircraft in every country where there was a manufacturing industry, but by July 1954 was forced to the conclusion that an aircraft meeting the ideal specification was just not being produced and few manufacturers were interested in the design of an aeroplane for which they could forsee a somewhat limited market.

At this particular time, of the 194 aircraft engaged in aerial work, 158 were Tiger Moths and over the next few years the number of this type increased to nearly 200.

Following the activities of the Committee already referred to, the Fletcher Aircraft Corporation of Pasadena became interested in the possibility of markets in Australasia and after consulting with the authorities and operators in New Zealand designed the FU 24 which, in practically all respects, met the specification drawn up in 1952. Design and prototype stages were accelerated and the first production aircraft became available within two years, the first models going into operation in 1955-56.

Unfortunately the urgent necessity to acquire additional equipment in the years 1952-56 had resulted in the introduction of Auster, Piper PA 18 and Cessna aircraft for both topdressing and spraying and the outlay of considerable sums on these types and their spares was, possibly, one of the greatest factors against standardisation on the single "ideal" type. Subsequently, the Auster Agricola and the Edgar Percival E.P.9 were produced as answers to the operators' specification, but these aircraft have not yet sold in quantity, largely because they were four to five years late and they are competing against established aircraft well entrenched in the aerial work industry.

The position now is that there are some half dozen types of aircraft well established in the aerial work field, each quite satisfactory in its particular role and it appears reasonable to assume that future increases in operators' fleets will be based on these types, with the possible exception of the DH 82 now, of course, out of production.

At the end of February 1959 the following were the principal types of aircraft used in aerial work operations: —

Tiger Moth DH 82	76
Piper PA 18A	63
Fletcher FU 24	51
Cessna 180	45
Auster (all marks)	22
DHC 2 Beaver	6
Helicopters	6
Miscellaneous types (including one DC-3, three Lockheed 18s)	11

It will be seen that the DH 82 still leads the field numerically and, while it was considered not entirely suitable for topdressing in 1952, it is now being hailed by many as the ideal sprayer for low level work. Certain hormone sprays (*e.g.* DNBP), insecticides in the organo-phosphorus group and defoliants must be sprayed at heights of 3 ft. to 10 ft. and pilots consider the DH 82 the most stable and safe aircraft for operation at these heights. It is not unlikely, therefore, that DH 82 equipment retired from topdressing may well (subject to airworthiness) be transferred to spraying.

Notwithstanding the general standardisation on such types as the FU 24, Cessna and PA 18A, operators are still seeking aircraft for special variants of existing forms of aerial work. In this connection interest is directed towards the Piper Pawnee, the Ag 2, the Ceres and, to a certain extent, the locally constructed but Australian-designed PL 11.

It will be noted that four "heavy" aircraft are included in the topdressing fleet. The suitability of the large aircraft for this type of aerial work has long been a subject for argument, the advocates claiming that DC-3s or Bristol Freighters operating from established airfields would have particular advantages where road transport charges for bulk fertiliser deliveries were high, where rural roads could be adversely affected by heavy vehicles and where the nature and extent of the terrain made operations unsuitable or hazardous for light aircraft.

A licence was granted in December 1955 for a DC-3 to be converted by one operator to carry 6 tons of fertiliser and to be used over a period during which the suitability of the aircraft could be evaluated.

Subsequently a second company was licensed to operate three Lockheed Lodestars and these, together with the DC-3 have been used under charter arrangements in many parts of the North Island of New Zealand.

Experience has shown that the so-called heavy aircraft are suitable for fertilising, and it is significant to note that the aircraft have achieved utilisation figures in excess of most light aircraft—the DC-3 averaging nearly 1,000 hours per annum during the period 1956-58.

Whether or not there is justification for an increase in the number of heavy aircraft is a matter of opinion at the present time. It can, however, be stated categorically that there is a place in aerial work for both the light and the heavy aeroplane with, perhaps, the former predominant.

The relative values of the fixed and rotary wing aircraft have also been examined in New Zealand, but no conclusions have been reached at variance with experience

gained in other countries. There is no doubt that the helicopter is an ideal vehicle for the delivery of sprays in certain classes of crops, particularly where low flying is necessary to counteract the possibility of damage arising from the drift of hormones.

Aerial topdressing, seeding, lime sowing and supply dropping are now well established in agriculture, and the techniques and operational procedures devised over the years have been largely stabilised. The same position does not apply in the aerial spraying field. The widespread demand for the aerial application of insecticides, herbicides, fungicides and rodenticides in the poison and deadly poison categories has necessitated a special study of the techniques to be adopted and the precautions and safeguards for pilots, ground staff and the public.

Where organic phosphates, dinitro phenols or classified deadly poisons are to be dropped, open cockpit aircraft with the hopper forward of the pilot location may not be used. Cabin aircraft with forward hoppers may be approved, subject to special tests to determine penetration of chemical.

This applies also to the application of the derivatives of fluoroacetic acid (*e.g.* 1080 and 1081) if these are applied other than as solid baits for rodent eradication.

One of the principal problems in both topdressing and spraying is the corrosive effects of fertilisers and agricultural chemicals generally on the metal components of aircraft. No totally effective preventive appears to have been produced, but a number of substances now available, plus frequent cleaning, act as a satisfactory palliative.

It is generally considered by the topdressing operators that the economic security of the industry has been assisted in no small degree by the system of licensing now in force. Air transport operators and aerial topdressing activities may be conducted only pursuant to a licence granted by the Air Services Licensing Authority, a statutory body of three members appointed by Government.

In respect of aerial topdressing the Authority grants an operator a licence applicable to a certain specified area or areas, and every endeavour is made to ensure that the capacity of operators is not grossly in excess of the actual or potential business in the area. The system of zoning adopted certainly does limit acute competition and assures operators of adequate business and utilisation of equipment.

Aerial spraying is not yet subject to licensing and the nature of control (apart from technical supervision) of this aspect of aerial work is presently under consideration.

DISCUSSION

After the lecture a number of questions were asked by members of the Society's Agricultural Aviation Group and the following is a summary of those questions and the replies received from Mr. Scott.

Reference was made to a statement in the paper by E. A. Gibson, published in the June 1958 JOURNAL, implying that overseas designers were incapable of thinking in terms of New Zealand's special requirements. Mr. Scott said that there was a time when operators in New Zealand had little success with overseas designers and could not get a topdressing aeroplane to meet local requirements. The ideal arrangement was considered to be one where operators could use ordinary materials, available locally, for repairs and replacements. This was understood to be permitted, to a certain extent, in the U.S.A. for aircraft operated under C.A.R.8. There were no design or manufacturing organisations in New Zealand, however, so the use of local materials had never been a major issue and orthodox practice still continued for agricultural aircraft.

Asked about a report indicating an estimated drop of 40 per cent in topdressing operations this year, Mr. Scott said that forecasts for agricultural activities for the year ended 31st March 1959 were unduly pessimistic. There had been a drop of 15 per cent in topdressing compared with the previous year, but there had been an increase of 60 per cent in spraying. The drop in topdressing was attributed to the effect of new taxation legislation on the farming community.

Replying to questions about C.A.A. recommendations and restrictions on the use of some spray materials, Mr. Scott said that even small quantities of organic phosphates and dinitrophenols could have a serious effect on pilots and handlers and the action taken by the Civil Aeronautics Administration was part of its safety programme. They had directed that for aircraft spraying organic phosphates and dinitrophenols the hopper should be aft of the pilot's position unless there was provision for (*a*) completely closing the top opening of the hopper and, (*b*) a filling point behind the cockpit. This requirement did not apply to cabin aircraft with forward-loading hoppers if tests proved that spray materials could not penetrate the cabin during filling. These requirements were not mandatory for chlorinated hydrocarbons but were recommended for them.

Asked whether there was a potential market in New Zealand for an aircraft which would also be suitable for the United Kingdom, bearing in mind the different terrain, Mr. Scott referred to the Appendix to his paper in which the main characteristics of the ideal agricultural aeroplane were outlined. If there was an aircraft suitable for use in the United Kingdom which met all these requirements, it should be equally suitable for New Zealand, but there was probably not a big enough market at this juncture to warrant the design effort and construction in quantity. Aircraft now is use, such as the PA 18A, Fu 24 and Cessna, were well entrenched with good spares and repair organisations and it would be difficult to alter this situation at present.

Some Aspects of Engine Noise*

by

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(Deputy Director, National Gas Turbine Establishment)

1. Introduction

Although a great deal has been said and written in the past eight years or so about engine noise, previous discussion has been very largely concerned with the physical aspects of jet noise and with the possibility of suppressing it by means of special propelling nozzles and ground mufflers. There is however another, and in general a better, way of dealing with jet noise, namely by using engines of low specific thrust, *i.e.* of low exhaust velocity and one purpose of the paper is to examine the possibilities of such engines. This leads to a consideration of blade noise from fans and turbines for as jet noise is reduced this tends to become the dominant factor. But first of all it is necessary to re-examine the nature of the noise problem.

The best introduction to this question of jet noise is to listen to some representative examples. The first group of noises selected for presentation and examination are those of some typical aircraft landing and taking off from London Airport; a second series compares typical aircraft noises at the same sound pressure level to bring out the subjective differences between them. The main things that strike one on listening to these records are:—

(*a*) the landing noise is as bad as the take-off noise or even worse,

(*b*) when the characteristic noises from different aircraft are compared at the same sound pressure level it is evident that subjective differences are very significant indeed. In particular there is a big difference between jets and piston-engined aircraft,

(*c*) with jet aircraft approaching the observer, especially on landing, there are indications of discrete frequencies with the suggestion of blade noise from the compressor intake.

The first point is straightforward and is easily confirmed by analysis. Figs. 1 and 2 show variations of sound pressure level with time at stations under the path of the aircraft landing and taking off and record the position of the stations and the height of the aircraft passing overhead. These graphs confirm that the maximum sound pressure levels under the paths of aircraft landing are at least as high as those under the take-off. The duration of the high noise level is less however. Note the second peak on the landing records of Super Constellation and Comet due presumably to the use of reverse thrust for braking. Note also that the maximum noise levels from the jets landing are reached when the aircraft is very nearly overhead and the noise patterns are nearly symmetrical about the zero time axis.

The second point brings a forcible reminder that noise is in the main a subjective phenomenon and that the sound pressure level which our instruments measure and the effect on human beings can be very different. This question is considered in some detail later.

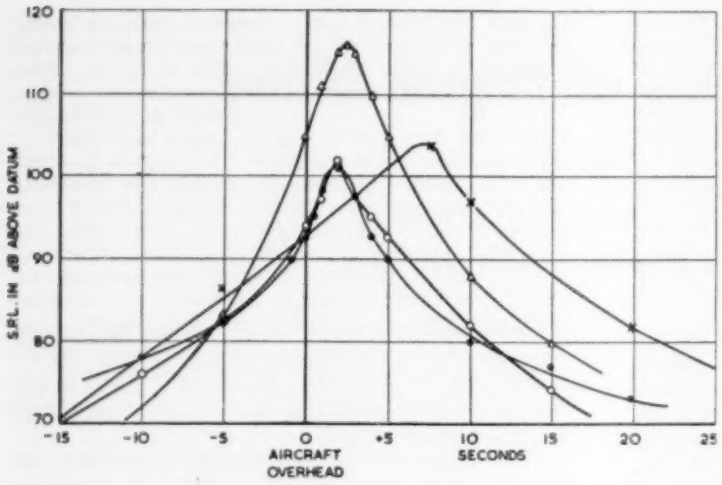
The third point, concerning intake noise, also needs fuller consideration, but reference to Fig. 8, which gives an analysis of various engine noises into narrow band widths, shows that discrete frequencies are indeed present in the noise radiated from the Boeing 707 to an observer in the front hemisphere. The peaks in the region of 2,800 cycles and 5,600 cycles are evidently the first and second harmonics of the first row compressor blade frequency from the four engines running at slightly different speeds. It is worth noting that although these discrete frequencies can be clearly heard and seen on the noise spectrum they make only a limited contribution to the total noise energy (100 db. out of 107 db.). By contrast the rearward radiation is essentially white noise.

2. Subjective Effects of Aircraft Noise

This question has recently been examined by Beranek[1] and his co-workers in the course of their thorough investigation of the noise of certain aircraft done on behalf of the Port of New York Authority and the American Public Health Association. The new units they have coined and the new scale of annoyance which they have proposed involve several assumptions and are unlikely to provide a final measure of noise quality, but Beranek's work represents a big step forward in the subject, and is undoubtedly of immediate value. The following paragraphs briefly summarise his argument.

The subjective quality that matters most in assessing aircraft noise is not loudness but annoyance. The distinction may seem to be a fine one but tests with a number of subjects confirm that it is real. The results are illustrated in Fig. 5, which shows the sound pressure levels at which noises in various octave bands were judged equal in annoyance to a reference sound (a pure tone of 1,000 cycles or a band of noise in the middle octave band 600 to 1,200 cycles). These comparisons were made using reference sounds at a range of

*A lecture given to the Society on 27th January 1959.



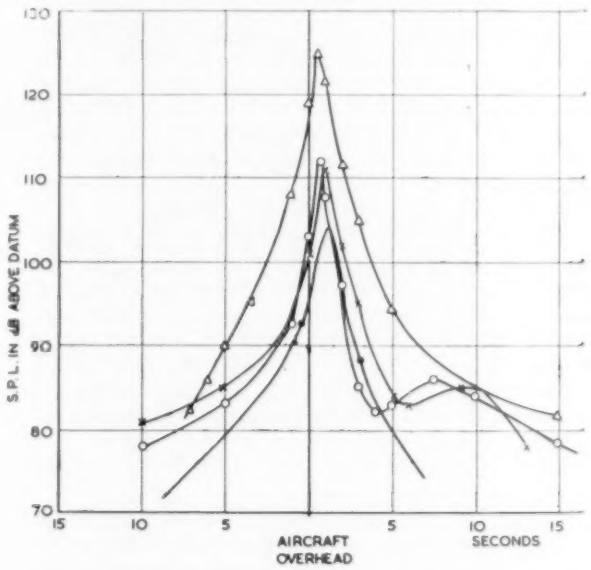
△ Boeing 707/120, 430 ft.
× Comet 4, 1040 ft.
○ Super Constellation, 472 ft.
● Britannia, 613 ft.

FIGURE 1 (*left*). Sound pressure levels of some aircraft taking off from London Airport, January 1959. Station for B.707 2½ miles from start of roll. Station for other aircraft 2 miles from start of roll.

FIGURE 2 (*below*). Sound pressure levels of some aircraft landing at London Airport, January 1959. Station for B.707 3,000 ft. from threshold of runway. Station for other aircraft 1,500 ft. from threshold.

△ Boeing 707/120
× Comet 4
○ Super Constellation
● Britannia

Height over microphone about 180 ft.



pressure levels. The contours of equal annoyance diverge appreciably from the contours of equal loudness especially at the high frequency end of the spectrum. These laboratory measurements form the basis of Beranek's scale of annoyance or "perceived noise" and the perceived noise level of a complex sound is calculated from its octave band analysis in a manner analogous to Stevens' well known method for computing loudness levels. The effect is, of course, that a noise with a large content of high frequency gets a high rating on the perceived noise scale, and vice versa.

Another factor which may influence human reaction to aircraft noises is that of endurance, and Beranek proposes, somewhat arbitrarily, that increased endurance be considered equivalent to increased S.P.L. (sound pressure level); thus doubling the endurance would require the addition of 3 db. to give the corrected P.N. (perceived noise) value. He assumes further that endurance varies directly as altitude, the correction applied is therefore to add to the P.N. rating $10 \log_{10}$ (actual altitude/datum altitude).

Beranek has also organised *ad hoc* tests on the subjective effects of various aircraft noises using one hundred observers and testing them by the method of paired comparisons, *i.e.* by playing over to the observers at various sound pressure levels the take-off noises of two different aircraft, and asking them to say which of the two they considered the more annoying. The aircraft noises used in this comparison included those of the Super Constellation, which provided the datum, Boeing 707/120, Comet 4 and Caravelle and they were adjusted to simulate indoor listening conditions. Table I summarises the results of this trial so as to bring out the qualitative differences between the different noises (*a*) in terms of the perceived noise unit as deduced from analysis, and (*b*) directly by the method of paired comparison. Both comparisons are given in terms of the difference between the sound pressure levels corresponding to equal noisiness; line (*vi*) for instance records that to produce the same degree of annoyance in the *ad hoc* tests the noise of the jet aircraft should have S.P.L. from 7 to 14 db. lower than the piston-engined aircraft. The table gives effective corroboration to the perceived noise scale since this gives results within 2 db. of those following from the direct comparative tests. The final conclusion is that for equal subjective effect the Caravelle, which has engines without noise suppressing nozzles, should have a sound pressure level 7 db. less than the Super-Constellation which provides the datum, while the Comet 4 and Boeing 707 with suppressing nozzles require differentials of 10 and 14 db.

It is clear that these differences are very significant, especially since the tendency is now to judge all new civil aircraft by comparison with the precedent established by the existing heavy piston-engined aircraft. They mean that jet-engined aircraft should operate at lower sound pressure levels than those to which people have become accustomed with propeller aircraft and that noise suppressing devices which involve a marked frequency shift may not give the expected benefit.

The use of the perceived noise scale is further illustrated by recalculating the results of Figs. 1 and 2 in P.N. db. This is done in Figs. 3 and 4. These figures are not advanced as fully representative comparisons between different aircraft but rather to give a general picture of the noise sequence experienced fairly close to the airport; it will be realised that the perceived noise units used in these figures, being instantaneous values, have no correction for endurance. Several features will be noted: the very flat curve for the Comet 4 associated with its high rate of climb, the noise maxima occurring in all cases after the aircraft has passed overhead, the fact that for all aircraft the noise maxima are greater at landing than at take-off and the high levels recorded on the Boeing 707/120.

An obvious consequence of the heavy weighting of high frequency noise in the P.N. scale is that atmospheric attenuation, which is greatest at short wavelengths, gains added importance. This point is illustrated in Table II, also quoted from Beranek, which records the effect of altitude on the perceived noise from a particular aircraft, the Boeing 707/120. It is seen that in this case the effect of air attenuation is roughly counterbalanced up to 1,600 ft. range by the endurance factor.

It will be appreciated that the perceived noise scale does not attempt to deal with the possible effects of discrete frequencies on subjective reaction to noise. This is evidently a difficult subject but one that may gain in importance in future.

TABLE I
SUBJECTIVE COMPARISON OF THE QUALITY OF VARIOUS AIRCRAFT NOISES MADE BY BOLT, BERANEK AND NEWMAN

Simulated indoor noise from the four aircraft is compared (a) in terms of the perceived noise unit as deduced from analysis (b) directly by the method of paired comparison.

	Boeing 707	*Comet 4*	*Caravelle*	*Super-Constellation*
(*i*) S.P.L. of simulated indoor noise db.	88	94	95	96
(*ii*) P.N. db. uncorrected	99	98	98	97
(*iii*) S.P.L. difference for equal P.N. db.	−10	−3	−2	Datum
(*iv*) P.N. db. corrected for duration	101	104	101	97
(*v*) S.P.L. difference for equal corrected P.N. db.	−12	−9	−5	Datum
(*vi*) S.P.L. difference for equal noisiness in *ad hoc* comparison.	−14	−10	−7	Datum

TABLE II
EFFECT OF ALTITUDE ON PERCEIVED NOISE FROM A BOEING 707 AIRCRAFT

1	2	3	4
Distance from aircraft (feet)	*Reduction by effect of distance (Square Law)* db.	*Square Law effect and air attenuation* P.N. db.	*Column 3 corrected for endurance* P.N. db.
400	Datum	Datum	Datum
800	6	8	5
1,600	12	19	13
3,200	18	33	24

3. The Jet Engine as a Noise Source

The jet engine has two main sources of noise, the jet itself and the blading; combustion noise can be neglected. Of the first of these two we know a good deal; of the second very little.

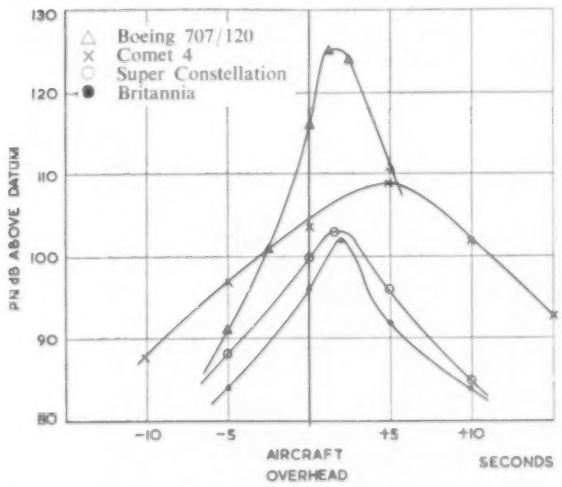


FIGURE 3. Perceived noise levels of some aircraft taking off from London Airport, January 1959. Station for B.707 2¼ miles from start of roll. Station for other aircraft 2 miles from start of roll.

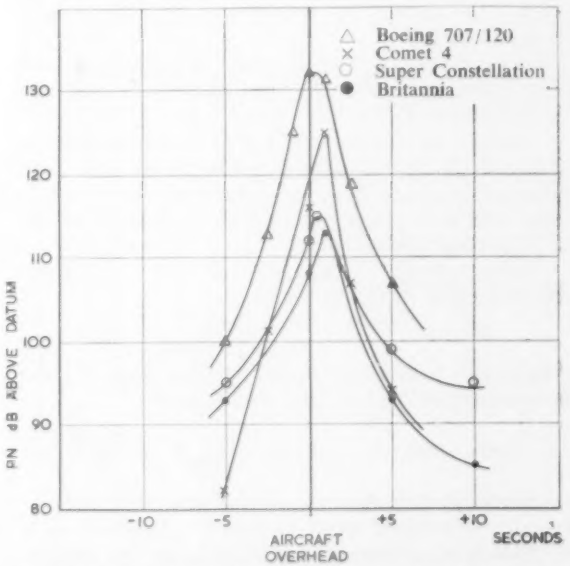


FIGURE 4. Perceived noise levels of some aircraft landing at London Airport, January 1959. Station for B.707 3,000 ft. from threshold of runway. Station for other aircraft 1,500 ft. from threshold.

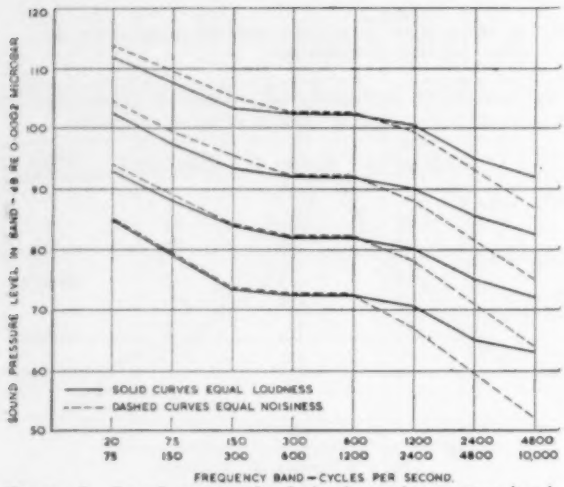


FIGURE 5. Sound pressure levels in the various octave bands to give (*a*) equal loudness (*b*) equal annoyance. From Beranek's study of aircraft noise.

Jet noise is in fact well enough understood to allow us to calculate it for untested or projected engines within reasonable limits, the basis for this being the well known theoretical work of Lighthill, the model work of various authors, and the full scale work of Greatrex and his colleagues at Hucknall. The basic relationships are:

Noise power of a jet = constant $\rho^2 AV^n$

where ρ = gas density, A = propelling nozzle area, V = jet velocity.

Here, for subsonic or just supersonic jets n is about 8. But if instead of total noise radiation we consider S.P.L. in the direction of maximum radiation 40° to the jet axis, the velocity exponent is about 9. The relevant jet velocity is taken to be that of the fully expanded stream.

Since thrust is given by ρAV^2 it follows that the static noise rating for a given thrust level will be a linear function of ρV^7.

Taking the numerical constants established at Hucknall we get a general expression for the noise rating of an engine based on ground level static conditions, 10,000 lb. thrust, and 800 ft. slant range at 40° to the jet axis as follows:

Noise rating $= 10 \log_{10} 1{\cdot}46 \times 10^{-10} \rho V^7$ db. above datum (0·0002 dynes/cm.^{2})

where ρ is in lb./ft.^{3}

V is in ft./sec.

Alternatively a measured sound pressure level E_m may be corrected to the standard conditions as

$$\text{Noise rating} = E_m + 10 \log_{10} \left[\left(\frac{x}{800} \right)^2 \times \frac{10{,}000}{F} \right]$$

where x = slant range to observation point in ft.

F = engine thrust in lb.

This is an approximate treatment but the simplification is useful in allowing us to make a general comparison of a wide range of engines under the static operating conditions which in fact give the highest noise outputs. The result is shown in Fig. 6 where noise rating is plotted for a variety of existing engines and some hypothetical ones against jet velocity, the dominant parameter.

The first conclusion is that subsonic turbo-jet engines of good economy have exhaust velocities at take-off which are normally in the range 2,000 ft./sec. ±100 and noise ratings between 117 and 119 db.; there is not much variation between them. Military engines and supersonic civil engines may have exhaust velocities of 2,300 ft./sec. with corresponding noise ratings of 122 db. At the other end of the scale ducted fan engines may work with jet velocity 1,200 ft./sec. and with noise rating of 103 db.

A similar comparison can quite easily be made for a flight condition, a suitable one being that corresponding to the early stage of the climb, with the engine still at take-off rating, altitude 500 ft., slant range as before 800 ft. and forward speed 150 kts. This is done by making the reasonable assumption that relative speed (jet speed less aircraft speed) is the relevant parameter so that a change in jet speed has a bigger effect. On this basis the difference between a jet engine of 2,000 ft./sec. jet velocity at take-off and a ducted fan of 1,200 ft./sec. goes up from 15 db. to 19 db.

So far we have considered only jet noise and this has indeed been the dominant factor, but the earlier survey of aircraft noise suggested that compressor noise also makes some contribution. This supposition is confirmed by Fig. 7 which shows the noise radiation from the intake of an engine running on the test bed with silenced exhaust. Corresponding maxima for jet noise for this engine would be 133 db. S.P.L. and 138 P.N. db. respectively, so that in subjective terms the intake noise is within 13 P.N. db. of the exhaust noise. This figure may give a rather exaggerated idea of present compressor noise level in that the high frequency component at least will be more confined by a flight intake, but it is clear that suppression of jet noise will tend to bring blade noise into prominence and will make it necessary in designing lifting fans and propulsive fans to take serious account of this factor.

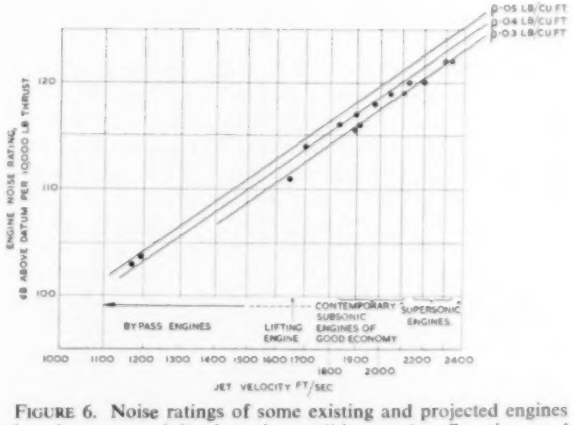


FIGURE 6. Noise ratings of some existing and projected engines based on ground level static conditions, take-off rating, and 800 ft. slant range.

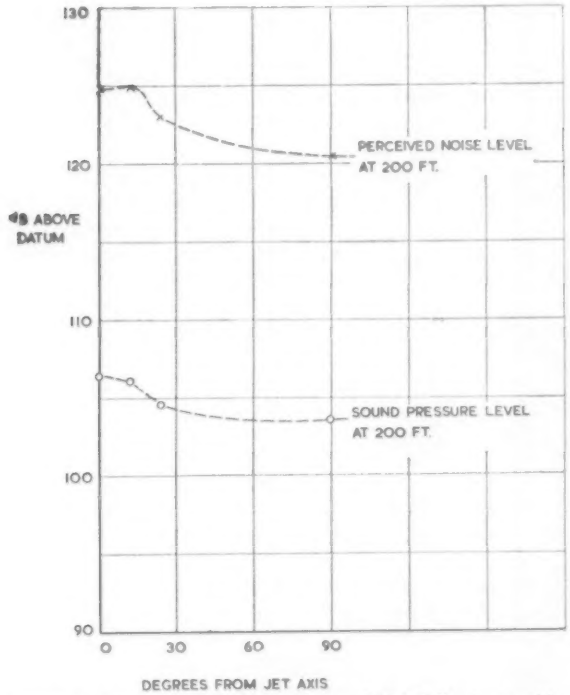


FIGURE 7. Intake noise from an engine of 10,000 lb. thrust with silenced exhaust. (Corresponding maximum jet noise 133 db. S.P.L. or 138 P.N. db.)

No general treatment of blade noise can be put forward in the present state of the art for neither propeller theory nor experience of ventilating fan noise can be expected to give much guidance as to what happens in the more controlled flow of a multi-bladed axial flow compressor. First indications are, however, that the analogy with the propeller is not very close. The noise output from propellers is dominated by the discrete tones associated with the harmonics of the blade rotation frequencies while compressor noise seems to be made up largely of (white) vortex noise

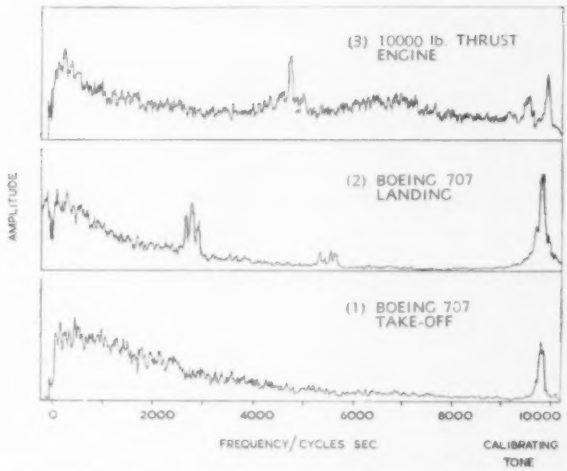
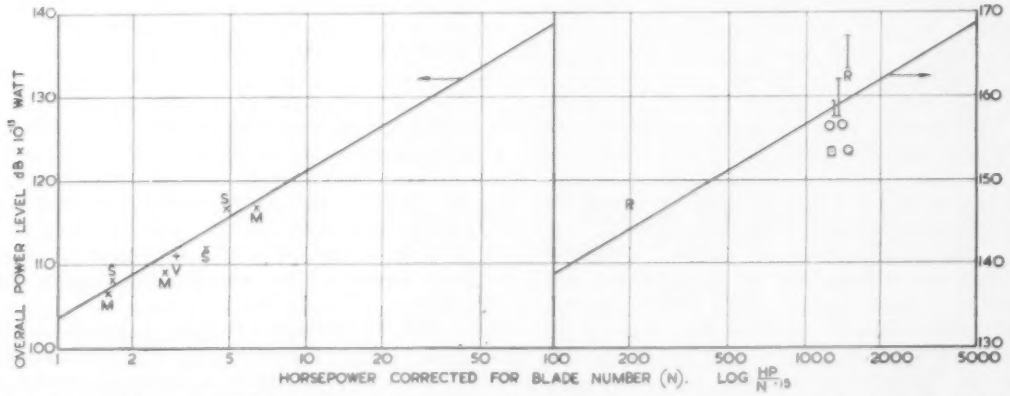


FIGURE 8. Noise spectra for various engine and flight conditions. (1) Boeing 707 take-off at London Airport 2·04 seconds after aircraft overhead. (2) Boeing 707 landing at London Airport 2·22 seconds before aircraft overhead. (3) 10,000 lb. thrust engine at take-off rating. Static conditions with test bed intake, and silenced exhaust.

arising in the blade wakes. This is well illustrated by Fig. 8(3), the noise spectrum from the intake of a contemporary engine, in which the discrete frequencies (1st and 2nd harmonics of the blade frequency, especially from the first row) though clearly shown are found to make only a limited contribution to the total noise energy. The band at 4750 cycles contributes 108 db. and that at 9770 cycles 96 db. to a total of 114.5 db. Hargest of N.G.T.E. has indeed found that Peistrup and Wesler's correlation[2] of ventilating fan noise, in which H.P. absorbed × (blade number)$^{0.15}$ is the controlling parameter, applies quite well to the vortex component from some aircraft fans and compressors (*see* Fig. 9). But the only certain comment at this stage is that the whole matter requires investigation to establish the effects of such obvious factors as blade Mach number and design, and to develop means for



S, Silentvane fan (10 blades). M, Multivane fan (60 blades). V, Ventilating fan (8 blades). O, Twin spool engine intake (53 blades). R, Single spool intake (37 blades). P, Typical control fan (42 blades). Q, Propeller (4 blades).

FIGURE 9. Vortex noise of some fans and compressors correlated in accordance with the Peistrup and Wesler formula.

suppressing compressor noise in case these should be needed.

Finally it is of some interest to consider how jet noise varies with thrust on a given engine. To a first approximation we can ignore the variation of gas density over the operating range and deduce that noise level will follow (thrust)$^{4\cdot5}$. This provides the convenient rule that noise reduction on throttling back from thrust F_{max} to thrust F is $45 \log_{10} \frac{F_{max}}{F}$ db. Fig. 10 shows that this need not be very far wrong. But if it is important to reduce noise to a greater extent on an aircraft having a margin of power available in the climb, this can be done by maintaining constant r.p.m. and opening a variable propelling nozzle so as to change the operating point to a lower pressure level, so getting a greater change in jet velocity for a given reduction in thrust. This effect too is shown in Fig. 10.

4. The Use of Engines of Low Jet Velocity to Reduce Noise

With jet noise determined by the eighth or ninth power of jet velocity the obvious way to reduce it is by cutting down this velocity. This could be done with straight jet engines by reducing cycle temperature and/or pressure ratio but doing so would not only give greater specific weight, it would also reduce engine efficiency and increase specific fuel consumption. In consequence the performance penalty for the noise suppression achieved would be a considerable increase in engine plus fuel weight. There is however, another way of reducing jet velocity (and specific thrust) which allows the inevitable increase in engine size to be offset by improved economy and hence a decrease in fuel weight.

This is to extract power from the main (high pressure) stream of gas in order to move an additional mass of air, the two streams being either mixed for passage through a common propelling nozzle or discharged separately. In this way it is possible to retain a gas generator of high pressure and temperature ratio, giving a good thermodynamic efficiency, and at the same time to realise the enhanced Froude efficiency of a jet of reduced velocity. This idea is by no means new, indeed it is as old as jet propulsion itself. Whittle's patent of 1936 envisaged a type of ducted fan engine and his 1940 patents describe thrust augmenters of what we would now call the rear fan type. In these an existing jet engine was to be used as gas generator, power being extracted in a free L.P. turbine having fan blades as integral extensions of the turbine blades. A number of these augmenters were designed by Power Jets Ltd. and one of them, the No. 4 augmenter, was the subject of a good deal of development and test work. The main emphasis in those days was on the achievement of increased thrust, and provision was made for an "after-burner" in the by-pass stream. Another wartime development, and a very notable one, was the Metropolitan-Vickers F.3 engine designed by Dr. K. Baumann in 1941; also of the aft-fan type with integral turbine and fan blades, this design had the

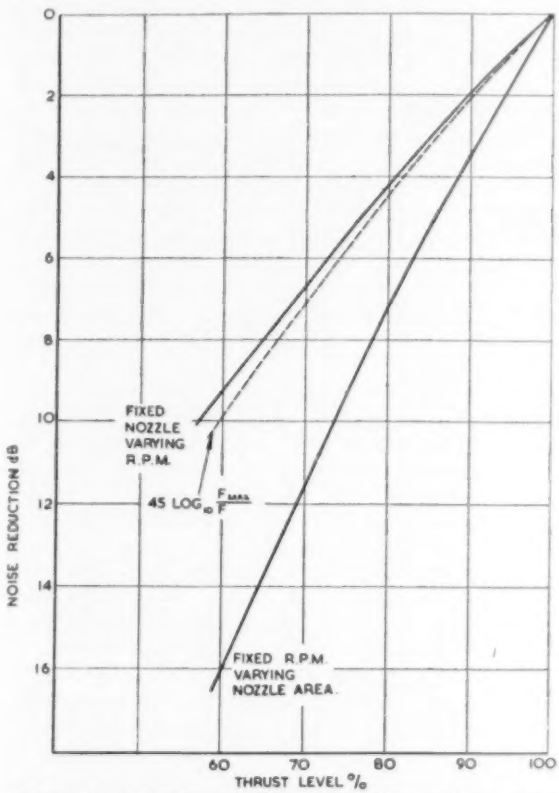


FIGURE 10. Reduction of noise of throttling back an engine. The full lines refer to a particular engine of 11·7:1 pressure ratio).

further refinement that the two stages were contra-rotating with consequent elimination of fan stator blades. This engine was bench-tested in 1943 giving an augmentation ratio of 1·65 and a specific consumption of 0·654 lb./hr./lb. The by-pass ratio was about 3·5.

The alternative arrangement of using a ducted fan both as a low pressure compressor for the main engine and also to provide an annular stream of air by-passing the high pressure engine system was considered soon after, and a variant of the Power Jets L.R.1 engine was shown with a geared drive from the main rotor to a two stage front fan. But this engine was never developed and the true pioneers of the front fan type engine are of course Rolls-Royce Ltd. whose Conway engine was the first of its kind. In the present context one must however state that the current Conways with their low by-pass ratios cannot be classed as quiet engines, their noise ratings being very similar to those given by straight jets. (For further information on the early history of by-pass and fan engines *see* recent article by D. L. Brown[3].)

The conventions which have grown up as to the nomenclature of these engines are unsatisfactory. Thus the term by-pass engine is habitually restricted to the Rolls-Royce product although the by-pass principle is in fact the essential characteristic of all these schemes. Similarly the use of the term ducted fan for engines having some arbitrary by-pass ratio whose "fans" may

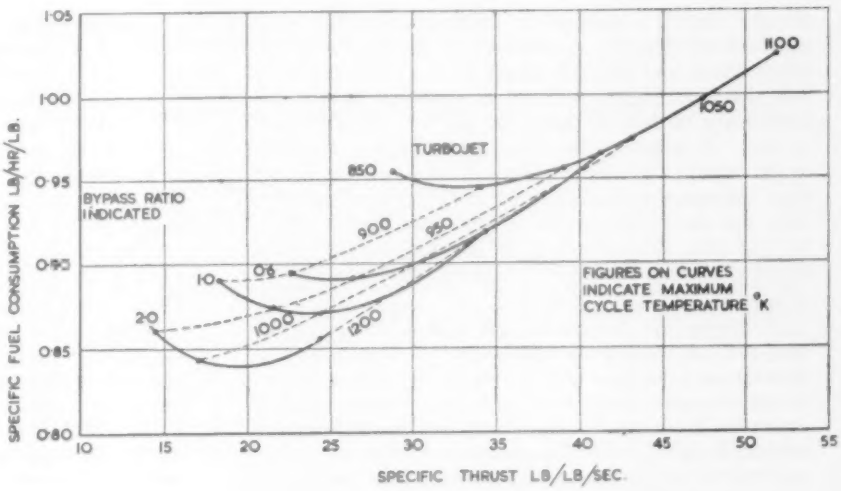


FIGURE 11. Cruising performance of a family of engines ranging from straight jets to 2·1 by-pass engines. Based on a flight speed of 500 kts. at 36,000 ft.

well compress the air to two atmospheres or so is open to some criticism. This difficulty will perhaps resolve itself in time; for the present I accept the name fan engine but prefer to use by-pass engine as the generic term recognising that to indicate noisiness one needs to know also either the by-pass ratio or the jet velocity. The other and more reprehensible misuse of the Queen's English is to call a silencer a device which so manifestly fails to achieve what the word promises.

The modern by-pass engine will be based on a gas generator of high pressure and temperature ratio, but if it is of the rear-fan type it will otherwise be very similar to the M-V engine of 1943. The only big difference in the contemporary rear fan engine by The General Electric Co. Ltd. is that the fan is a transonic one giving the whole temperature rise required in one stage. Front-fan by-pass engines have changed more drastically in that the development of cooled turbines and of highly loaded compressor stages have made possible the important simplification of eliminating the geared drive for by-pass ratios up to say 2: 1. Thermodynamically there is no real difference between the two types and the conflict between them will be decided in terms of engine weight, nacelle drag and other factors. But the front-fan layout has been more thoroughly studied in this country and has fewer unknowns about it, so the performance figures which follow are based on this type.

The performance obtainable from modern by-pass engines under subsonic cruise conditions is illustrated in Fig. 11 which shows specific fuel consumption at cruise plotted against specific thrust for a family of (hypothetical) engines. (The assumptions are: flight speed 500 kts., pressure ratio 12, fan efficiency isentropic 86 per cent, H.P. compressor efficiency polytropic 89 per cent, combustion efficiency 98 per cent, combustion pressure loss 5 per cent, turbine efficiencies 88 per cent, no thermodynamic advantage from mixing of exhaust streams.) The data of Fig. 11 are based on engine thrust uncorrected for external drag; if this factor is taken into account assuming a podded installation with (nacelle area)/(engine area) = 1·9 and taking drag coefficient based on wetted area = 0·006 the improvement in s.f.c. offered by a 2:1 by-pass engine against a simple jet falls to 7 per cent. In such an installation the overall efficiency (thrust horse power/rate of fuel consumption) is about 22 per cent. If the take-off rating of the two engines is adjusted to give the same ratio of thrust at T.O./(thrust-drag) at cruise the noise ratings work out as shown in Table III.

In the engines considered it has been assumed that the by-pass and main streams are mixed to give a simple jet at a substantially uniform velocity. In practice such complete mixing may not be worthwhile and we may use propulsive jets having a velocity gradient across the diameter; alternatively we may have separate jets in which the annular by-pass jet will be at a rather lower velocity than the central core. The noise characteristics of such composite jets have not yet been properly explored and it is not possible at present to say how far they will differ from those of uniform jets, but the differences are likely to be small.

It is worth pointing out that the reduction in jet noise which the by-pass engine offers would seem to be sufficient not only to enable a large jet aircraft to meet the tentative requirements of comparability with heavy piston-engined aircraft but to set an improved standard. Assuming an installed take-off thrust of 80,000 lb. and a likely frequency distribution for a large engine we get a perceived noise for the flight condition of 95 + 3 + 9 = 107 P.N. db. This is less than the accepted value for present practice and would be further reduced, on an aircraft of reasonable climb performance, by throttling back before reaching inhabited areas. This sum makes no allowance however for blade noise.

Finally there is the question of engine weight. This obviously depends very much on the skill of the

TABLE III

	Cruise T_{max}	*T.O.* T_{max}	*Thrust at T.O. / (Thrust-drag at cruise)*	*Noise rating static* db.	*Noise rating at climb condition* db.
Jet engine	950°K	1,150°K	4·14	119	114
2:1 by-pass	1,150°K	1,300K°		104	95

designer and is a subject on which it is hard to give comparative figures. A ducted fan engine giving the same cruise and take-off thrust as a simple turbo-jet will always be bigger in linear dimensions but will not necessarily be heavier since the size of the H.P. unit is less. In practice the difference in weight is likely to be small, one recent estimate showing that if no drag allowance is made, *i.e.* for a fully buried installation, the ducted fan engine is 4 per cent lighter, while for full drag allowance it is 2½ per cent heavier. The increase in engine size may however introduce a further penalty in installation weight.

Summing up: the advantages of the ducted fan engine are (*a*) improvement in noise rating sufficient to provide the basis for a real solution of the noise problem in subsonic aircraft (*b*) worthwhile improvement in fuel economy; its drawbacks are (*c*) the need for higher cycle temperatures than the comparable turbo-jet so that it may be harder to achieve the necessary reliability, (*d*) greater size and complexity and probably some increase in weight.

The balance between these factors which decides what is the best engine for any particular task will evidently depend on various factors: aircraft speed, size, and performance in the climb and the noise level which can be tolerated, and no rigid conclusion can be drawn. But it does seem clear that for subsonic flight it is intrinsically better to achieve whatever noise level is required by using engines of low jet velocity rather than by taking simpler jet engines and suppressing their noise at the cost of performance penalties. Certainly this conclusion is fully supported by the only detailed study of the problem which has been published, namely that by Pearson and Fitzgerald[4]. They considered the case of a short-to-medium range airliner and showed that whereas 10 db. noise reduction by noise suppressing nozzle would add 6 per cent to direct operating cost the same reduction obtained by means of special engines would give a similar reduction in direct operating cost. For long range aircraft the case for using special jet engines would be reinforced and the optimum by-pass ratio would be greater.

All that has been said refers to subsonic flight and to runway-based aircraft. Supersonic civil aircraft redouble the difficulties for the following reasons:

(*a*) The supersonic civil aircraft is essentially a large aircraft and its low speed performance may be poor. It will therefore need a high thrust level at take-off.

(*b*) Fig. 7 shows that engines for supersonic propulsion tend to have high noise ratings.

(*c*) While there may be scope for using by-pass engines at the lower supersonic speeds it is inescapable that these engines are penalised, when operating at high intake temperatures, because material and engineering considerations make it impracticable to use the cycle temperatures which thermodynamic considerations would demand.

It follows that the supersonic civil aircraft constitutes a new noise problem to the solution of which the engine of low jet velocity is less readily applicable.

The V.T.O. aircraft is also a special case because in the civil context this only makes sense if it can operate direct from centres of population. Here the limits of acceptable noise are likely to be rather lower than in the suburban areas surrounding present airports and they will be defined at stations much closer to the noise source. In terms of jet velocity this may mean that while 1,200 ft./sec. is all right for runway-based aircraft, 600 ft./sec. may be the corresponding figure for V.T.O. At this condition it is evident that fan and turbine noise are likely to be dominant.

5. Outstanding Problems

The conclusion that the engine of low velocity offers a complete solution to the problem of jet noise from subsonic civil aircraft and does so without imposing any performance penalties, must be qualified by the posing of two questions:

(*a*) What are the factors controlling noise radiation from turbo machinery, and how far can this noise be reduced by design methods?

(*b*) How are we to define and measure the nuisance value of engine noise in which discrete frequencies are present and significant?

We also have in the offing the new and difficult problem:

(*c*) What can one do about the noise of the supersonic airliner?

ACKNOWLEDGMENTS

The noise records referred to in the paper and analysed in Figs. 1-4, 7 and 8 were taken by W. B. Jenkins and M. J. Porter. Their help and that of other colleagues at N.G.T.E. is gratefully acknowledged.

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TECHNICAL NOTE

The Prediction of Three-Dimensional Stress Concentration Factors

by

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TWO-DIMENSIONAL STRESS concentration factors may be obtained more quickly and simply than the corresponding three-dimensional factors, either by experiment or mathematical analysis. It would be convenient to obtain information, for varying geometry in the two-dimensional case of a particular type of stress raiser, *e.g.* a shoulder, groove or hole, and use this either to predict the three-dimensional stress concentration factors or to extend the range of existing three-dimensional results. Clearly a comparison is only possible if the three-dimensional stress raiser embodies a plane of symmetry (which gives the geometry of the similar two-dimensional stress raiser), and if the loading conditions can be reproduced in both the two- and three-dimensional cases. The latter requirement restricts the correlation to the stress concentration factors obtained in tension and in bending. The three-dimensional torsional loading system has no plane of symmetry which can be simulated in two dimensions.

NOTATION

D diameter of shaft
d small diameter of shouldered shaft, or minimum diameter of grooved shaft
r radius of shoulder, or root radius of groove
K_3 stress concentration factor for three-dimensional stress raiser
K_2 stress concentration factor for the corresponding two-dimensional stress raiser
$R=(K_2-1)/(K_3-1)$

PETERSON'S CORRELATION

Peterson[1] has obtained a range of stress concentration factors for cylindrical shouldered shafts in tension by substituting the results for a shouldered plate[2] in the following relationship:—

$$\frac{K_{2S}-1}{K_{3S}-1}=\frac{K_{2G}-1}{K_{3G}-1}=R \qquad . \quad . \quad (1)$$

Where (*see* Fig. 1):—

K_{2S} stress concentration factor for a thin shouldered plate obtained by experiment[2]
K_{3S} stress concentration factor for a shouldered shaft with the same geometry as the plate
K_{2G} stress concentration factor for a notched plate by calculation using Neuber's theory[3]
K_{3G} stress concentration factor for a grooved shaft (similar to the plate) by calculation using Neuber's theory[3]

Clearly, since the Neuber notch and the shoulder fillet are not geometrically similar, equation (1) implies a value of R which is sensibly independent of geometric shape, for notches of the same severity. The parameters h/d and r/d (Fig. 1) are by implication measures of notch severity. The ratio $(K_{2G}-1)/(K_{3G}-1)$ for grooved cylinders and the corresponding notched plates is plotted in Fig. 2(*a*), the stress concentration factors being taken from Peterson[1]. These graphs are derived from Neuber whose analysis is exact only for infinite bodies. The difference between real finite boundaries and an infinite hyperboloid is likely to be greatest for small values of r/d. The graphs shown in Fig. 2(*a*) (chain lines) are inconsistent between themselves, whereas the algebraic expressions of Neuber would give a continuous variation with the two parameters r/d, h/d. This inconsistency is not significant since it would imply an accuracy of plotting of one per cent in the original graphs.

The ratio $(K_{2S}-1)/(K_{3S}-1)$ for shouldered shafts and the corresponding plates is also plotted in Fig. 2(*a*) (full lines), the stress concentration factors in this case being taken from two- and three-dimensional photoelastic tests[4][5]. For a shoulder in a shaft or a plate the stress concentration factor is generally less than three, in which case the ratio R is very sensitive to errors in K_2 and K_3. For $h/d=0{\cdot}15$ it has been calculated that a three per cent error in the values of K_2 and K_3 can result in an error in R between ten per cent (for $r/d=0{\cdot}025$)

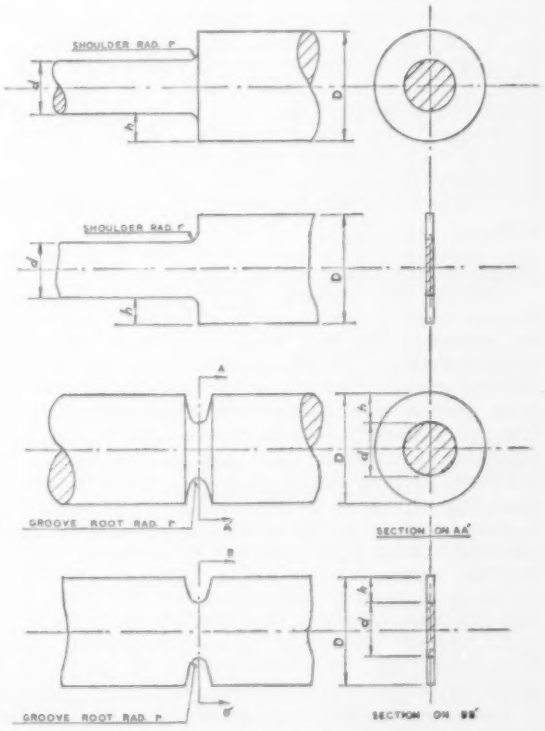

FIGURE 1.

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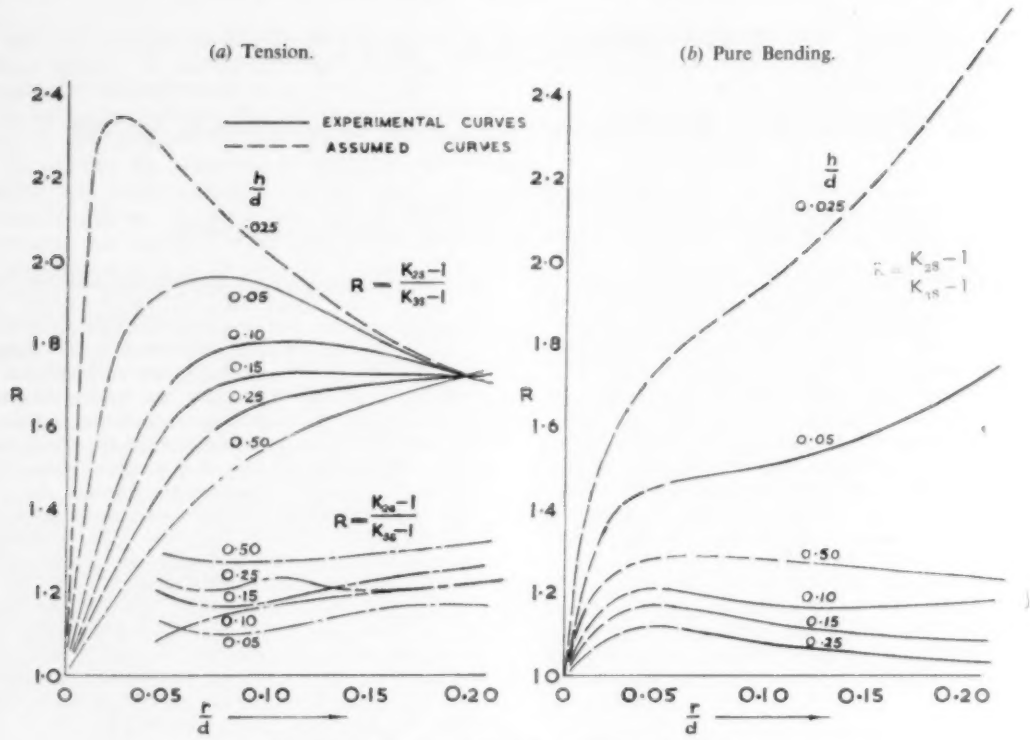

FIGURE 2. Ratio R against r/d for constant values of h/d.

and 16 per cent (for $r/d=0{\cdot}2$). Notwithstanding this the corresponding values R for groove and shoulder are significantly different in the range covered by the experiments, and it must be concluded that the assumptions of Peterson are not borne out by experiment. His method would always predict three-dimensional shouldered shaft results which are significantly higher than the true values. In this particular case Peterson's shouldered shaft curves are based on two-dimensional photoelastic results by Frocht[2] which are known to be low so that the error is less than it might have been.

LIMITED EXTRAPOLATION

Although the sweeping assumptions implied in the Peterson approach are unjustified and possibly dangerous, the continuous and well behaved nature of the experimental curves given in Fig. 2 indicates the possibility of inferring three-dimensional results from two-dimensional stress concentration factors within well defined limits. Tests at University College, London, show that it is impossible to correlate stress concentration factors for completely different forms of loading. However, for a given type of loading the relationship $(K_2-1)/(K_3-1)=R$ may be used, if the three-dimensional stress raiser embodies a plane which is a plane of symmetry for both the body shape and the loading system. Under these conditions K_2 must in general be obtained for the complete two-dimensional test range and K_3 for a number of points spread throughout the corresponding three-dimensional range. The behaviour of the values of R calculated for these points shows whether it is possible to make accurate predictions of the stress concentration factors over the whole three-dimensional range.

The procedure is illustrated by its use in extending the range of the stress concentration factors for shouldered shafts, obtained in (*a*) tension and, (*b*) pure bending, from photoelastic models[4]. The three-dimensional results are plotted as full lines in Figs. 3(*a*) and 3(*b*), the range of the experiments being indicated by the shaded area. It was found to be impossible to extend the range of the three-dimensional tests to the desired limit as $r/d\rightarrow 0$, owing to the practical limitations on maximum casting size and minimum shoulder radius. However, large scale two-dimensional models could be made and a complete range of two-dimensional stress concentration factors[5][6] obtained for both tension and pure bending. As $r/d\rightarrow 0$ it is not unreasonable to suppose that the two- and three-dimensional factors will tend to the same limit and that R will tend to one. In Figs. 2(*a*) and 2(*b*) an extrapolation of the shouldered shaft curves (shown as broken lines) to this limit does not conflict with the general trend. Since these curves have been drawn from experimental data, their use to predict the three-dimensional stress concentration factor K_{38} is valid provided the extrapolation is correct. Furthermore it can be shown that relatively large errors in R, for $|R|<2$, operating on accurate values of K_{28} produce small errors in K_{38}. The "predicted" values obtained by operating on the shouldered plate results of References 5 and 6 are plotted on Figs. 3(*a*) and 3(*b*), as broken lines.

The three-dimensional photoelastic stress concentration factors for both the loading conditions have been plotted keeping each of the parameters h/r, h/D, h/d and r/d constant in turn, to give four families of curves, one of which is given in Fig. 3. Since the same information was plotted in each case corresponding points in each family must be consistent and the final smooth curves

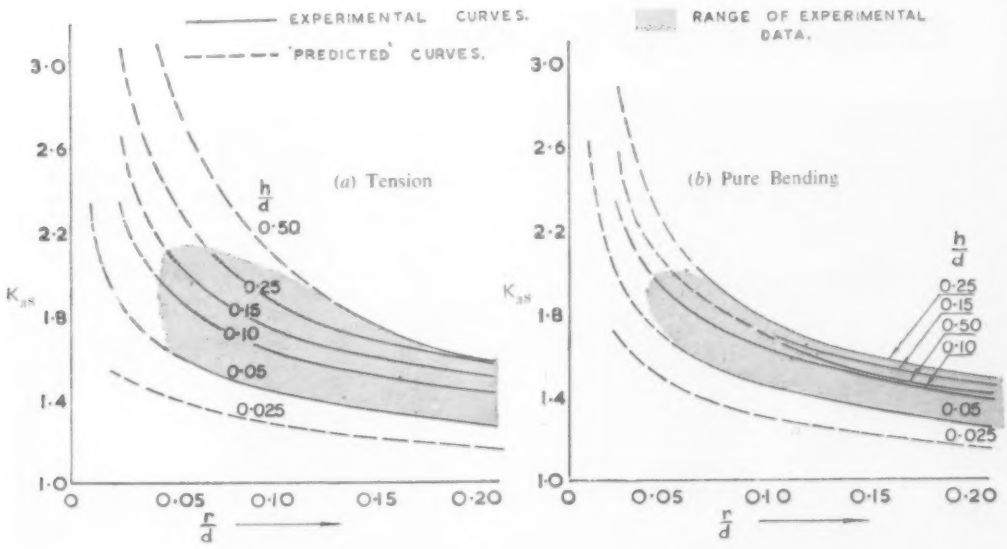

FIGURE 3. S.C.F. K_{ss} against r/d for constant values of h/d

were drawn to satisfy this condition, while paying due regard to the experimental points. This "compatibility" condition represents a severe test on the accuracy of the experimental results and the choice of a suitable family of curves permits of a limited extrapolation of the three-dimensional results. A comparison of the "predicted" and the "extrapolated" results for tension and for pure bending is given in Table I.

In the test range the maximum divergence between the results is 3½ per cent and this is not considered to be significant in view of the possible inaccuracies of the two methods used to obtain K_{ss}.

TABLE I

Tension $h/d=0{\cdot}05$, $r/d=$	0·05	0·025	0·0167	0·0125
extrapolated	1·61	1·83	2·06	2·23
predicted	1·61	1·82	1·99	2·17
$h/d=0{\cdot}025$, $r/d=$	0·1	0·05	0·025	0·0125
extrapolated	1·26	1·40	1·53	1·66
predicted	1·28	1·38	1·52	1·62
Pure Bending $h/d=0{\cdot}05$, $r/d=$	0·05	0·025	0·0167	0·0125
extrapolated	1·67	1·94	2·17	2·35
predicted	1·65	1·95	2·18	2·39
$h/d=0{\cdot}025$, $r/d=$	0·1	0·05	0·025	0·0125
extrapolated	1·26	1·44	1·68	1·97
predicted	1·29	1·45	1·67	2·03

CONCLUSIONS

It is only possible to infer three-dimensional stress concentration factors from two-dimensional stress concentration factors in certain restricted circumstances. The three-dimensional stress raiser must contain a plane of symmetry, the shape of which specifies the geometry of the corresponding two-dimensional stress raiser. No correlation is possible between completely different types of load, and the mid-plane of the solid body must also be a plane of symmetry for the loading system in order to provide an equivalent loading system in two dimensions. The latter condition excludes torsion problems, but does not restrict the method to cases of axial symmetry only. Examples of the type of loading which satisfy these conditions include tension, bending, shear in the plane of symmetry and combinations of these systems. In every case a sufficient number of three-dimensional tests must be made to determine the appropriate shape of the R curve.

The best use of the method appears to be in "filling in" information when a reasonable amount of three-dimensional background data is already available. For example, it is likely that a study of the appropriate R curves for simple cases would enable rapid and accurate predictions of the effects of combined loading on three-dimensional stress raisers to be made over a complete range. A small number of three-dimensional tests would be required as a check for each type of stress raiser.

It is thought that there is a considerable danger in applying this method to a completely new problem, when a lack of previous information might easily provide a plausible but fictitious result.

ACKNOWLEDGMENT

The author's thanks are due to Mr. C. Snell, Head of the Photoelastic Laboratory at University College, London, for his encouragement and assistance.

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Graduates' and Students' Section

(Courtesy of Pan American Airways)

October's visit to London Airport will include the Pan American maintenance area, where we hope to see the Boeing 707.

Visits

The September visit is on Monday the 14th to the Ministry of Supply Aircraft and Armament Experimental Establishment at Boscombe Down, near Salisbury, Wilts. In October the date is Wednesday the 14th and the place is London Airport. This is really two visits in one. First we will be shown the Ministry of Transport and Civil Aviation Control Tower and then travel to the South West Maintenance Area on the far side of the airport where Pan American will be our hosts. While not their main base, a considerable amount of work is tackled here, and if our visit is well timed we may get a close look at the Boeing 707. Applications for both these visits should be made early to the Hon. Visits Secretary, Mr. A. R. M. Pickering, "Riverside", Vicarage Walk, Bray, Berks.

Film Show—Wednesday 23rd September

An exceptionally fine programme of films has been booked for our first Film Show:—

"Fairey Rotodyne"—A colour film of the design, construction and development of the world's first Vertical Take-Off air liner.

"Antarctic Crossing"—A newly released colour film of Dr. Vivian Fuchs's Commonwealth Trans-Antarctic Expedition.

"Man in Space"—A new documentary by Walt Disney in colour describing the not too distant future.

"Aeronautical Oddities"—A hilarious film of Man's more weird and haphazard ideas for flight with and without wings.

We are continuing the practice which was started last year of serving free refreshments in the form of coffee and biscuits at 7 p.m., with the lecture or film show beginning at 7.30 p.m.

Lecture on 7th October—Lunar Probes

Mr. J. E. Allen, who will lecture to the Section about lunar probes at 7.30 p.m. on Wednesday, 7th October, is Head of the Aerodynamics, Projects and Assessment Department of the Avro Weapons-Research Division. After gaining a first class B.Sc. of London University in 1941, he joined the R.A.E. where he worked on aero- and hydro-dynamic research. Later he did flight-test studies on flying boats at Helensburgh and further research at Felixstowe. On returning to the R.A.E. he studied the ballistic and aerodynamic problems of V-bomber weapons before joining A. V. Roe and Co. Ltd.

This year Mr. Allen was awarded the J. E. Hodgson Prize by the Society. A council member of the British Interplanetary Society, he is well-known for his papers, articles and lectures on astronautics.

In his lecture Mr. Allen will describe the lunar probes which have been launched up to the present time and will examine the design and guidance requirements for such vehicles. He will also discuss the kinematics of moon trajectories and will extend his talk to cover manned as well as unmanned lunar vehicles. His lecture will be illustrated by slides, models of lunar-probe vehicles and an orbit model of the earth and moon made specially for this occasion.

THE LIBRARY

Reviews

HELICOPTER DYNAMICS AND AERODYNAMICS. *P. R. Payne. Pitman, London, 1959. 442 pp. Illustrated. 84s.*

This is a volume which cannot fail to be of interest and value to the practising helicopter engineer and more advanced students of the helicopter art, and the author must be congratulated on having presented so much material in one volume. The printing is excellent and there are some 300 explanatory drawings and graphs, the latter in the main presenting the basic material from British and American reports and articles in the Technical Press.

Proof reading must have been a task of the greatest magnitude, on the whole well carried out, although there are a few typographical errors and misprints in formulae which have been tested.

The first three chapters on General Aerodynamics, Induced Aerodynamics, and Fundamentals of Rotor Dynamics are excellent in every way, and presented in a way which a reader not conversant with earlier literature can understand. Exception must be taken however to the statement made so early as p. 3 that it is normal to neglect compressibility for all performance and aerodynamic calculations, except control angle to trim. Anyone who has had to analyse authenticated helicopter performance trials must know that on types with high rotational speed and high forward speed, estimates of power required are not satisfactory in the absence of corrections for compressibility (and drag rise due to blade stalling).

Chapter 4 in its entirety is devoted to one of the author's pet theories that the demerits of the conventional fully articulated rotor can be avoided by incorporating an elastic stiffness into a centrally located hinge. The effect is exactly the same as that of providing high offset at the flapping-hinges, except that the effective offset of the elastically restrained hinge varies with rotor speed.

The remaining chapters deal with Flapping Stability, Rotor Vibration, Ground Resonance, Control Loads and Vibration, Blade Flutter and Blade Flexing, all of which subjects are only to be found in the reports by specialists. All are presented in a form making them excellent introductions to these difficult and somewhat abstruse subjects. A minor criticism is that only hovering stability is discussed.

The main criticisms which can be levelled at the work reside in the chapter on Performance. As the author so rightly points out, the reader of a volume such as this, with so much mathematical treatment, is always careful to check formulae by deriving them for himself, yet, although unfamiliar parameters are used, the performance formulae are given in many instances without any clue as to how they are derived. This can be very disconcerting to the reader, and if the necessary clues are given elsewhere in the book, cross references should invariably be given.

In the past, a wide variety of helicopter performance methods have been published, and they have been a source of much confusion. Where, as in the present volume, yet another method is presented based on totally new thrust, torque and power coefficients it is important that the relationship to other well known formulae should be given. It is particularly confusing to find C_T, C_Q and C_P in the present work used for new coefficients defined by the author, when the same symbols are already in general use for corresponding coefficients referred to disc area, tip speed and density. These symbols by the way are never used, as stated by the author, for the earlier British coefficients referred to half density. Here a separate set of symbols, usually T_c, Q_c and P_c, was always used so that no confusion can be caused by the present general acceptance of G, C_Q and C_P referred to density.

The author's arguments with respect to the value of n in the $(1+n\mu^2)$ correction for radial flow are quite beside the point. Professor Bennett's mean value of $n=4{\cdot}65$ is based on sound approximations, is universally accepted, and gives the right answer when the mean blade drop is corrected for forward speed, blade stalling and compressibility.—R. N. LIPTROT.

THE SILVER DART. *H. Gordon Green, Brunswick Press, Canada, 1959. $4.95.*

This book is a biography of J. A. D. McCurdy who is called "Canada's First Pilot." He was the first Britisher to pilot an aircraft over British soil or, to be precise, over British ice, for his first flight was made during the winter off the ice of Bras d'Or Lake, at Cape Breton Island, N.S. on 23rd February 1909. Actually Canada's first pilot was F. W. Baldwin who, since he flew in the State of New York on 12th March, 1908, was not only Canada's first pilot but the first British subject to fly an aeroplane, if you ignore Henri Farman who was then still a British subject.

This book, which on the dust cover is indicated as being "published under the authority of the National Co-ordinating Council for the Golden Anniversary of Flight in Canada," is disappointing because it is inaccurate. It makes improper claims, none of which are necessary to the appreciation of the pioneering work by Mr. McCurdy. If this book is authorised as indicated, it is a shame that the above-mentioned Council did not carefully check the facts and dates mentioned in the book. If it is a book which gets a wide circulation in its present form, many will be misled and even more arguments about who flew first and in what will be started. One feels that there are enough arguments right now.

Mr. McCurdy was one of the five members of the Aerial Experiment Association which consisted of Alexander Graham Bell, Glenn Curtiss, F. W. Baldwin, J. A. D. McCurdy and Thomas Selfridge. This small group of people worked for a short period from 1st October 1907 until 31st March 1909, financed by Mrs. Bell. During that time they built several powered aircraft, some of which flew at Hammondsport, New York, and one (Silver Dart) at Baddeck, Nova Scotia.

It is clear from the book that the author has had full access to the reports and minutes of the Aerial Experiment Association, but in spite of that, errors as mentioned have crept in, and although many would be apparent to anyone with aeronautical knowledge some of them may not be so obvious, and I am indebted to Mr. J. H. Parkin for pointing out to me some of the more refined discrepancies.

For instance, on page 41 the implication is that their first aeroplane, "Red Wing," was a conversion from a

glider, which it was not. A few pages later it states that the engine that powered the "Red Wing" was not used on other aircraft, whereas in fact it was used on two other aircraft—"White Wing" and the "June Bug."

On page 56 it refers to a Curtiss engine as the "first water-cooled aircraft engine with copper-jacketed cylinders." It is not clear whether this is intended to imply that this was the first water-cooled engine used in aircraft which happened to have copper-jacketed cylinders, or whether it is claiming to have been the first water-cooled engine with copper-jacketed cylinders: in fact, it was neither. The Wrights used water-cooled engines from the beginning, and the "Antoinette" engine with copper jacket flew earlier.

Inaccuracies in the book occur in connection with the flights before the Canadian Army on 2nd August 1909. The book talks about five flights, including turns. Actually only four flights were made without turns, and considerably shorter distances than that indicated in the book.

On page 44, it is stated "the development of the aileron was entirely that of the associates, and was one of the really great aeronautical advances of the age." This refers to the group of five people forming the A.E.A. mentioned before. This statement is open to considerable criticism. It may well be that the idea of the aileron was independently conceived by Bell, and worked on and developed by Baldwin and McCurdy, but it is wrong to give the impression that other people had not independently worked on the aileron. John Stroud advises me that there is evidence that Blériot used ailerons on his "Libellula" in 1907 and also on his "Blériot VIII" which he flew in 1908. Farman also claims the invention of the aileron and used ailerons on his first aircraft in 1909. It has also been claimed that Santos-Dumont was the first to use ailerons in 1906. The "Antoinette" which attempted to fly the Channel in 1909 had quite normal looking trailing edge ailerons. It was on 13th May 1908 that the first ailerons designed by the A.E.A. were flown. These were movable extensions of the wing tips and were on both the upper and lower wings. They were more in the form of the early version of G. T. R. Hill's wing tip controllers than the present-day aileron.

As for the French name for the "aileron," this is in itself evidence that it originated in France. However, the book under review states that some time after the flight of 1908 McCurdy visited Farman who on being told of this new gadget exclaimed in French "Why, it is a little wing." This is taken to indicate that the aileron was news to Farman, but in view of the remarks above, it could be just as well confirmation by Farman that he knew all about it.

Another claim made by the book on page 45 is that the A.E.A. originated the tricycle landing gear. I think there is no doubt that this is quite wrong because there are several examples of tricycle undercarriages before the work of A.E.A. was undertaken.

Later in the book, there are other inaccuracies, one being, for instance, on page 157, where it is indicated that in 1928 McCurdy formed the Reid Aircraft Company in Montreal. I am advised that this was formed by W. T. Reid and the Board of Directors did not even include Mr. McCurdy.

On page 159 it states that Gordon McGregor has been President of Trans-Canada Air Lines since 1945 whereas he has held that post only since 1948. This is a small error, but indicative of the fact that one must query almost every figure in the book.

Apart from the inaccuracies exampled, the book is loosely written and full of irrelevancies which could have been replaced by some of the information omitted, particularly in connection with McCurdy's later work in Canada. If detailed technical information is desired, it is suggested that the paper *The Evolution of the 'Silver Dart'* by J. H. Parkin in the "Canadian Aeronautical Journal" of February 1959 should be consulted. This is far more valuable than the book under review. In addition, the paper entitled *Aircraft manufacturing in Canada during the First Great War* by K. M. Molson in the same issue of the "Canadian Aeronautical Journal" gives interesting information on some of the later activity of Mr. McCurdy which is dealt with inadequately in the book under review.

Efforts are being made in Canada to limit the circulation of the book until it has been revised, and it is hoped that this will be applied to any sales of the volume in the United Kingdom.—B. S. SHENSTONE.

SQUADRON HISTORIES R.F.C., R.N.A.S. and R.A.F. 1912-59. *Peter Lewis. Putnam, London. 208 pp. Illustrated.* 30*s*.

By commissioning Mr. Lewis to produce this book Putnams no doubt intended to add yet another to their excellent series of historical books on military aviation. Potentially this could be a very valuable contribution to R.A.F. history for a handy reference book on the facts and figures of the R.A.F. Squadrons has long been conspicuous by its absence. It is sad therefore to record that this book falls lamentably short of its predecessors both in fact and presentation.

The body of the book comprises so-called histories of each Squadron from No. 1 to No. 695. Actually the title of the book is a misnomer for each squadron is dealt with by presenting an agglomeration of facts in chronological order with, in most cases, no narrative to convert the data into a history. This is followed by thirteen appendices of which some are apposite while others appear to bear no relation to the subject matter of the book and would have been better omitted. One appendix lists the named or titled squadrons. Of the 139 "names" in the list 65 are obsolete, unofficial or inaccurate (our friends in Northern Ireland will not readily accept Ulster's downgrading to county level). The two most important appendices are those giving a list of squadrons by aircraft type and listing squadron code lettering. Neither have been cross-checked adequately with the body of the book; for example, under A. W. Meteor NF.11 it is stated that No. 139 Squadron was equipped with this type but on looking at the "history" of 139 Squadron no reference is made to the Meteor NF.11; quite rightly, for 139 has not flown fighter aircraft since 1919! The other appendix, while giving a comprehensive list of the squadron code letters used between 1938 and 1950 (and in some cases up to the present time) does not give the dates the letters were used by the respective units, thereby setting a trap for the unwary, a trap into which Mr. Lewis himself seems to have fallen in several of his squadron histories.

There are no photographic illustrations but Mr. Lewis has given us eight pages of fighter squadron markings in colour and here he has excelled himself. It is quite the most comprehensive and accurate collection of these markings yet to appear in print.

Unfortunately, the text falls down very seriously on two counts, those of factual accuracy and consistency. So serious are these faults as to render the book largely valueless as a historical record. It is no exaggeration to say that this book is riddled with errors of fact. Your reviewer is well aware, as is Mr. Lewis, that certain facts

of squadron history are open to dispute and may, in fact, never be established but to these he does not refer. For example, it matters little whether one record shows a squadron was disbanded on 31st July 1947, while another record gives the date as 1st August 1947. But it does matter in a work of reference of this sort when it is stated that a squadron was transferred to India when in fact it went to Egypt. And this is no isolated "howler." While time would not permit a close scrutiny of the whole book to determine the number of errors contained, in order to be as fair as possible to readers and to Mr. Lewis a sample batch of fifteen consecutive squadrons was taken at random part-way through the book and subjected to a close but by no means exhaustive examination. In those fifteen squadrons nineteen errors of fact were discovered and that score excludes both petty discrepancies of date and aircraft mark number and also obvious typographical errors. In addition, as many statements again were found which, while not factual errors, by reason of their place in the context of the histories were misleading. Inaccuracies on this scale call into question the accuracy of all the facts between the covers: one is forced to suspect every statement until it has been verified from another source.

In addition, the book is inconsistent. No standard layout is adopted to cover the presentation of the squadrons; some squadrons contain references to one or two of the many bases used and a few random mentions of commanding officers, others do not; some record numbers of enemy aircraft destroyed or tonnage of bombs dropped while others omit even references to the type of operations carried out. An example of inconsistent checking is shown by the fact that no fewer than three different squadrons, Nos. 3, 67 and 234, are credited with being the first to use the Sabre F.4. But most inconsistent of all, perhaps, is the arbitrary allotment of space to the different squadrons. For example, No. 276 Squadron, an Air/Sea Rescue unit whose total life span was four years is given seventeen lines, whereas one of Bomber Command's oldest and most illustrious units, No. 10 Squadron, with thirty-five years' service, is squeezed into thirteen lines.

Mr. Lewis appears to have made an honest and sincere attempt at a task which to many would be just a chore and to others a task too immense to contemplate. But it is only too obvious that he has put together a welter of details without any set plan of presentation or without that checking, double-checking, cross-checking and re-checking of his facts that is so necessary in the compilation of this kind of work.

It now behoves Putnams speedily to produce a second edition in which the vast majority of these errors and inconsistencies have been eradicated by the diligent checking of the facts with those authorities, both official and unofficial, who specialise on this subject. Then, and only then, can this book be put on the library shelf as a useful and accurate work of reference.—J. D. R. RAWLINGS.

COME NORTH WITH ME. *Bernt Balchen. Hodder and Stoughton, London, 1959. 318 pp. Illustrated. Index. 21s.*

In the days of Colonel McCormick, of the Chicago "Herald Tribune," twisting the lion's tail was a good clean honest game—or at least as honest as anything else in politics. Since those days, the British have not become any more sensitive to the general criticisms of their American cousins, but they do maintain certain reservations about national conduct during the last war. A country which had to stand alone for eighteen months may with reason believe herself entitled to fair consideration—above all by her allies. But unfortunately this view is not held by Bernt Balchen.

In *Come North with Me* Balchen has marred an interesting account of his not undistinguished life by the penultimate chapter, describing the part Balchen played in helping his native Norway against the Nazis. Unfortunately he came up against a line of British government officials who, rightly from their point of view, but wrongly from Colonel Balchen's, would not agree to his plan for starting air operations on behalf of the Norwegian patriots. That this was due to a Norwegian agreement with Washington not having been communicated to any British authority is clear from the text, but in his impatience to get started, Balchen was only interested in effects and not in causes. To him the British were the arch-obstructors, and the fact that aeroplanes were urgently needed on other fronts, meant nothing.

Other allegations are that the British were charging the Norwegian Government a fee for each refugee brought out of Sweden by the Norwegians with their own equipment (p.262). One top-ranking British Officer is made to sneer "As a matter of fact, is not Norway one of our colonies?", and there is an allegedly verbatim conversation between Balchen and Air Chief Marshal Sir Trafford Leigh-Mallory, commander-in-chief of Allied Air Forces. The C. in C. is recorded as having expressed his embarrassment that the R.A.F. commander had refused to carry on supply-dropping during the long hours of daylight in Norway, and to have shown himself uninformed and obstructive throughout. Incidentally American phraseology is attributed to Leigh-Mallory which is in line with such inaccuracies as "Bovington" for Bovingdon, "Farnsborough" for Farnborough and the location of U.S.A.F.H.Q. at "Widewing, a suburb of London." On page 283 Balchen records that eight destroyers given by America to Britain had been subsequently sold by us to the Russians. The only good thing the British ever did during Balchen's war was when B.O.A.C. gave him office accommodation in Stockholm.

Balchen's record of himself is by no means all blunder and bravado, although at least one of his stories does not agree with that of another authority. For instance, Balchen's explanation of his failure to fly across Antarctica with Lincoln Ellsworth in 1934 is very different to the version given by Lincoln Ellsworth in his book "Beyond Horizons." Nevertheless his accounts of Amundsen's and Byrd's expeditions to the North Pole, bush flying in Canada and his experiences generally in the late 1920s and early 1930s make the first part of the book interesting reading.

An aviator who has flown in so many parts of the world as Balchen ought to have recognised immediately his obligation to promote friendship between all those nations who seek it. That he has done great harm to Anglo-American relations through publishing his book in America cannot be doubted: even if the British publishers see fit to eliminate the offensive remarks in subsequent editions, the major harm has already been done.. It is deplorable that Balchen could not have exercised the balance and skill in his writing of which he is evidently capable in the handling of aeroplanes.—JOHN GRIERSON.

KNIGHT OF THE AIR. *Maxwell A. Smith. Cassell, London, 1959. 228 pp. 18s.*

Another book about Saint-Exupery will probably seem unnecessary to many, yet some will probably agree that there is still scope for a proper interpretation in English of the life and writings of this remarkable man. Unfortunate-

ly, this latest study—which first appeared in America about three years ago—is only a doubtful improvement on its predecessors. From the point of view of a reader in the English language who has any knowledge of flying, Saint-Exupery's misfortune is that he was an intellectual, a poet and one of the great literary figures of his time. This has meant that his writings about flying have been translated and he himself has been written about by people with little or no understanding of his vocation. Dr. Maxwell Smith is apparently no different so that the translations he quotes, or makes himself, and his references to almost every aviation aspect of Saint-Exupery's career do not ring true to the technically-informed reader. The fact is that French, like every other language, when used by a writer to describe, even incidentally, any specialised activity—such as aviation—in which he is well informed, cannot be translated with satisfying realism into another language except by somebody who is familiar with that same activity in both languages. Saint-Exupery's writings are, of course, of much wider scope and significance than aviation alone but, because his best-known works have a flying setting, translations of them lose some of their impact from being technically inaccurate and lacking in authentic expression.

Readers interested in Saint-Exupery's life in aviation will be disappointed in the sketchy treatment given in this biography to the various phases of his airline career. In addition to a more factual account of his many interesting experiences as an airline pilot, test pilot, "record flyer" and operational pilot in a French Air Force reconnaissance squadron in 1939-40 and 1943-44, one would have liked to have heard more about his adventurous time as Aeropostale's station superintendent at Cap Juby in 1927-29 and his notable work as operations manager of Aeroposta Argentina in 1929-31. We are told about a few isolated incidents of those days but always in the context of his writings—which were after all only a spare-time activity at this time—and in such a vague and incomplete way that no convincing picture of the man or his life emerges. Indeed, the reader gains the impression that Saint-Exupery was always an absent-minded eccentric entirely preoccupied with his philosophical interests. This may have been to an increasing extent true of the last few years of his life, when like so many Frenchmen he was a frustrated exile from his over-run homeland, but it can hardly apply to his earlier days.

Typical of this book's tendency to treat everything on a vaguely etherial artistic plane and its failure to concern itself sufficiently with realities are its confused descriptions of Saint-Exupery's crash in the Egyptian desert in his Caudron Simoun in December 1935, while he was making an attempt on the Paris-Saigon record and the account of his death over the Mediterranean in 1944. No light is thrown on the crash in the Simoun. Contemporary news reports said that Saint-Exupery accidentally overflew the Nile Delta while in cloud at night and hit high ground about 90 miles east of Cairo. According to other—and probably more accurate—accounts the crash occurred in the Western Desert, 125 miles West of Cairo although there is no high ground in that area. We are no wiser after reading Dr. Smith. The story given of Saint-Exupery's death is that which has appeared several times before although, as Miss Babington Smith indicated in a letter to the *Sunday Times* on 19th April 1959, it is probably inaccurate. A more careful search in allied and enemy records might modify the details significantly.

The author of this book quite rightly emphasises that Saint-Exupery has been widely acclaimed as a visionary and philosopher. It was, indeed, these qualities in a talented writer and poet, who also happened to be a professional pilot, which made him famous in his lifetime. However, in this reviewer's opinion, his long-term reputation will rest much more on the fact that his three best-known books—"Night Flight", "Wind, Sand and Stars" and "Flight to Arras"—were among the first true literature of the air. He was perhaps the first great writer who was also an airman. It may be difficult for the English reader to appreciate this today, unless he can also read French, but the position will change if, and when, his writings are adequately translated. There will then also be scope for a more convincing biography than Dr. Maxwell Smith has been able to produce.—PETER W. BROOKS.

ROCKET PROPELLANTS. *Francis A. Warren. Chapman and Hall, London, 1958. 212 pp. Illustrated. Index. 52s.*

There is certainly room for a good book on rocket propellants, a field in which there have been such great developments as have made possible the beginnings of the "space age" in which we find ourselves. Whether a really satisfactory one can be written may be questionable, in view of military security restrictions, and also perhaps commercial secrecy in a field which is keenly competitive in the U.S.A. While making all allowances for these difficulties, however, the reviewer found the volume disappointing.

The eleven chapters of the book, including the bibliography of four to thirty references appended to each, have an average length of under twenty pages. Their titles are "Propellant Systems", Propellant Ingredients", "Solid Propellant Manufacturing and Processing", "General Performance Characteristics", "The Burning of Propellants", "Ignition and Igniters", "Solid Propellant Rockets", "Liquid Propellant Rockets", "Safety", "Evaluation and Quality Control", and "The Future of Propellants". Thus it may be seen that the field of the book is that of both solid and liquid propellants, and that besides the propellants, their ingredients, manufacture and mechanisms of reaction, it attempts some coverage of rockets and their performance. The longest chapter is that on Solid Propellant Manufacture and Processing, and discusses the solvent, solvent-less and ball processes of making colloidal propellants, processing of "cast" colloidal propellants, and the wide field of composite propellants, all in twenty-five pages of which one is devoted to "Inspection". The compression of so much material into so short a book, and such short chapters, is obviously liable to result in some inadequacies of treatment.

An example of the results of this degree of compression may be taken from the chapter on Propellant Ingredients where the manufacture of nitro-glycerine—a major ingredient in many important propellants—is described by one sentence "Nitro-glycerine is manufactured by the nitration of glycerine, using a 50/50 mixture of nitric and sulphuric acid". It is hard to believe that any useful purpose can be served to anyone by such an account of the manufacture, the methods for which are not subject to military security, nor (except perhaps the costs) to commercial secrecy. The next paragraph, discussing diethylene glycol dinitrate, says "It is manufactured by a method similar to that for nitro-glycerine, using glycol in place of glycerol" which not only suffers from the same inadequacy, but is wrong since obviously glycol dinitrate would be the product, not diethylene glycol dinitrate. The book contains several other statements which might mislead.

The chapter on "Safety" relates to practice based on American regulations for manufacture (inter-building distances, building constructions, permissible electrical

equipment, etc.) and transport. It has no appreciable value for manufacture or transport in other countries; for Americans engaged in propellant manufacture it is far from listing adequately the legal documents which impose safety requirements, or indicating whence they may be obtained; while for the general reader a photograph entitled "Protective equipment used by an explosives operator" and showing someone in fire-resistant overalls, wearing a hood and face shield, standing beside a water deluge nozzle while observing through another shield some process monitored also by a "heat detector", can only give a misleading impression of the general working conditions in the majority of propellant processing operations.

Unfortunately, these are not rare and isolated instances, but are almost typical of the treatment throughout the book. It is to be hoped that a more recommendable volume on the subject may become available before too long.—G. P. SILLITTO.

DYNAMICS OF FLIGHT—STABILITY AND CONTROL. *Bernard Etkin. Chapman and Hall, London, 1959. 519 pp. Illustrated. 120s.*

It is notable that though our bookshelves are now beginning to groan under the weight of the encyclopaedic age of aeronautics, few have written a small book and none a very large one on what is traditionally called stability and control. Doubtless many persons eminent in this subject, after looking wryly at the immense sprawl of its "literature", have asked themselves "Can I drive a straight road through this morass?", and have decided not to try. As for the encyclopaedic group of experts, their meeting would be very likely to break up in disorder on the question of notation, or on whether the Laplace and Fourier transforms are blessed or not. Let me therefore praise Professor Etkin of Toronto for his coolness and courage in surveying his road, for his competence in keeping it straight, and for his industry in driving it through to the end. His book has a clear form and a weighty but easily carried content; it is, as such things go, easy to read because he has obviously enjoyed writing most of it; and he writes it very well.

A work of this sort is lost unless it pursues one main object to the exclusion of all others. Professor Etkin's aim is to assemble a bag of mathematical tools for those who have to take the behaviour of an aircraft to pieces and put it together again. His bag contains, of course, more than one kind of tool, but he has the wit and wisdom to put them in one bag, not two, thus over-riding most of the arguments that have retarded the development of the subject in its later stages. To its early workers in this field an aircraft was no more than a fascinating tangle of derivatives; they worried at the differential equations of its disturbed motion, studying its normal modes with the saving graces of such approximations as luckily turned up. But the coming of automatic control put a new face on the matter. It seemed that the aircraft could also be regarded as a complex of transfer functions busily engaged in converting input into output. Was this a new revelation, or only a new and possibly quicker way of getting at the old answers? Many people, especially older ones, did not much like the look of the aeronautical block diagram; they liked even less the sound of the indicial and impulsive admittances, the gains and noise and frequency responses, which describe the way it works. Professor Etkin propounds the truth of this matter. These tools are complementary and co-operative. Put them in one bag, use each in its proper sphere, and you will do a cleaner and quicker job of work than if you insist on using one to the exclusion of the other. Moreover he puts his argument in the right order. Everything depends on getting a real grip of the general equations of motion. One cannot unfortunately have too much of these. Professor Etkin gives his reader the whole classical works, and then, in his chapter on Mathematical Aids, grounds him thoroughly in the practical aspects of the various operational transforms. This is as it should be.

There is an inevitable drift to greater complication as Professor Etkin's exposition proceeds. He knows something about instructional brinkmanship, the art of bringing the reader confidently to the verge of difficult knowledge and leaving him to take his cheerful plunge into formidable reference lists. For instance, we are warned that it is not good enough to represent the turbulent atmosphere by a few discrete gusts, and are given a glimpse of the strange statistical country governed by the power spectral density analysis. Again, knowing that missiles cannot escape him, he turns a difficult corner by giving a simplified analysis of the properties of one of them, the cruciform configuration. And finally there is a third class of tool that is almost too big to be got into his bag. Some of his sets of equations occupy a page of print. There they are, and if they cannot be simplified there is only one thing to do about them, whatever the addicts of simple approximations may say. The reader is given some elementary instruction in the nature of analogue and digital computers.

The stability derivatives, in one form or another, are always with us. They drag in their wake, on one quarter, almost the whole of aerodynamic theory, and on the other a mass of experimental technique. This is the only matter on which I am inclined to fault Professor Etkin. He finds himself in a quandary from which there is no neat exit. He cannot say nothing about this complex of data on which his tools must work; and if he deals faithfully with it his book might be doubled in length. His sketch of what the derivatives amount to reflects the boredom of one who has not left himself room enough to manoeuvre.

There is one omission which though deliberate must evoke regret. This is a treatise on the tools of the behaviour analyst; behaviour itself, though it is frequently used for illustration, is a secondary matter. All the same it would be of the greatest interest to make a critical survey of the changes in behaviour, as dictated by speed, all the way from the glider to the slender supersonic shapes. But this, Professor Etkin would justly say, is another story, and another book. Even our up-and-coming young scientists cannot have everything; this book gives a section of them more than enough to be going on with.—S. B. GATES.

COMBUSTION AND PROPULSION PROBLEMS. *Edited by J. Fabri, A. H. Lefebvre, O. Lutz and M. W. Thring. Pergamon Press, London, 1958. 614 pp. Illustrated. 140s.*

This volume contains the papers presented at the Third Colloquium arranged by the AGARD Combustion and Propulsion Panel, together with the discussions on these papers, and four other papers presented at an earlier panel meeting. Unlike the papers presented at the first two Colloquia, which were intended as general reviews of particular aspects of Combustion, the papers now presented cover a very wide range of propulsion subjects, while some of them describe only single lines of specialist work. As a result, few readers will be interested in more than a small proportion of the whole series.

The section on noise, however, is in the best AGARD tradition, with excellent review papers by Sanders and

North (N.A.S.A.) and by Richards (Southampton University). An accompanying paper by engineers of the S.N.E.C.M.A. deals in great detail with the methods and cost of silencing engine test cells, and the psychological effect of noise on testers.

The papers on combustion fall into two groups—those dealing with theoretical and laboratory studies, and those dealing with practical problems. The subjects of the academic papers include jet mixing, the interactions between flames and shock waves and flame stabilisation in a boundary layer. A paper describing small scale work on flame stabilisation by the opposed jet technique is followed by a description of tests on the S.N.E.C.M.A. Reheat System in which the flame is stabilised behind transverse jets carrying five per cent of the total air flow. On the practical side Levine (N.A.A.) describes, unfortunately with little detail, the use of a thin diametral segment of a combustion chamber of a liquid rocket engine for the study of "screaming" instability, and the use of an electrically heated tube for measurements of the transfer coefficient between a tube and a liquid coolant at heat flows up to 10 B.t.u./in.^{2}sec. The paper by Penner (C.I.T.) and others gives details of the various theories of the burning of composite solid propellants, but it is evident that these are by no means final.

A paper by Childs (N.A.S.A.) describes all the ways in which design of the combustion system is made more difficult by the behaviour of the rest of the engine—awkward velocity profiles at the compressor outlet, for example: one would have hoped that this defensive attitude of the combustion engineer should by now have vanished and that it should be possible to balance the difficulties inherent in compressor, combustor and turbine design to produce an *integrated* optimum engine design.

A paper by Spalding is a brave attempt to present the reaction rate theory in a form acceptable to both academic and practical men. The subsequent discussion suggested that his simplifications were too simple for the theoreticians, while practical men still consider his statements too complicated: perhaps they may yet be the foundations of a useful compromise.

On the subject of future progress in propulsion, Lombard's paper parallels the one he presented recently to the Royal Aeronautical Society, while the conclusion of a theoretical paper from O.N.E.R.A. on ram-rockets is that they are usually inferior to the appropriate rocket/ram-jet combination. In the final section, on Aerophysical Chemistry, there is a general paper on radiant heat transfer to a hypersonic vehicle, a detailed theoretical study of convective heat transfer to a solid surface with mass transfer and chemical reaction, a theoretical study of the vibration, dissociation and ionisation of atoms and molecules behind a really high temperature shock, and a paper on some special aspects of magneto-hydrodynamics. None of these four is suitable reading for beginners in these subjects.—S. L. BRAGG.

BALLISTIK. *R. E. Kutterer. Vieweg, Brunswick, 1959. 304 pp. Illustrated. DM 38. (In German.)*

In this third edition of Kutterer's work the previous versions have been thoroughly revised and extended to give adequate cognizance to the recent work on high-speed aerodynamics and rocket missiles which has modified the theories and techniques of classical ballistics.

Although the three parts of this book, and their subsections, are written with this point of view in mind, the two parts which show most clearly the impact of modern technologies are the section on "secondary" ballistics, which deals with the rotation, gyration and localised aerodynamic behaviour of projectiles, and the last part of the book, which is devoted to an overall, if cursory, account of rocket technology, considering the rocket as a ballistic missile.

Other sections which are of interest to physicists and aerodynamicists are those dealing with the rapid measurement of time, pressure, motion and related phenomena. Descriptions are given of the use of interferometry, the Kerr cell, Faraday shutter and image converters, for high-speed photography. Also of interest are the descriptions of unconventional weapons such as the electric cannon, the ram-rocket missile and the "high-pressure pump" gun.

Essentially this is a well-written, comprehensive textbook for the ballistician. Each section starts logically with the concepts of "classical" ballistics. Implications of modern research are examined and the classical analyses modified. The mathematical treatment is clear and within the scope of the science graduate. Numerous worked examples are given. If comments on some recent progress are necessarily brief, nevertheless bibliographical references are given to the original papers and the text is altogether well documented and illustrated.

The physicist and aerodynamicist will also find much of interest in the sections noted. The exposition is refreshingly clear and well set out and there is an author index. Altogether one has the impression that a great deal of ground has been covered in the comparatively short space of 300 pages.—F. A. BLAKE.

Additions to the Library

Cooperation Between Airlines: Economic and Political Aspects. L. H. Slotemaker and D. Goedhuis. **Prospects of Future Air Traffic Development Within the European Common Market.** B. Björkman, Centro per lo Sviluppo dei Trasporti Aerei. 10 pp. each. The first item consists of a paper each by a Director of K.L.M. and by the Professor of Air Law at Leyden. The second is by the former Secretary of the Air Research Bureau in Brussels.

Einführung in die Theorie der Strömungsmaschinen. A. Betz. G. Braun, Karlsruhe, 1959. 272 pp. Diagrams. DM 36. *To be reviewed.*

General Procurement Information on Guided Missile Programs, Rockets and Target Drones. U.S. Central Military Procurement Information Office, Office of Secretary of Defense. U.S.G.P.O. 1959. 28 pp. 1s. 5d. American missiles with the names of contractors and subcontractors responsible are listed.

Logic in Elementary Mathematics. R. M. Exner and M. F. Rosskopf. McGraw Hill, London. 1959. 274 pp. 52s. 6d. The object is to help the reader to understand how mathematics is communicated, and this is probably the first time an attempt has been made to introduce logic at the stage of elementary mathematics. Intended for secondary school or the first two years of college, the material is presented in such a way as to be accessible to those whose mathematical training includes the standard elementary courses but who lack any formal course in logic. Applications include solution of a linear equation, solution of a quadratic equation, definition of order, definition of absolute value, definition of a limit, and so on.

National Airport Plan—1959. U.S. Federal Aviation Agency. U.S.G.P.O. 1959. 504 pp. 42s. 9d. (H.M.S.O.) A "reference only" item giving details of airports (divided into "for Air Commerce" and "for General Aviation") in the United States. Statistics are given for each state including estimates for required development in the current and three subsequent years.

Raffaele Giacomelli. E. Pistolesi. l'Aerotecnica. 1957. 4 pp. No price. A reprint from *l'Aerotecnica* of a memoir of Giacomelli, the prolific Italian writer on aeronautical history, who died on the 13th December 1956. A chronological bibliography of his writings is included.

Regards neufs sur l'Aviation commerciale. Henry Laile. Editions du Seuil. 1959. 302 pp. Illustrated. 15s. A "popular" paper-back in a series entitled "Peuple et Culture." Profusely illustrated with photographs and drawings and with a bibliography for further reading.

Report on the Cost Structure of the Indian Airlines Corporation 1959. Stephen Wheatcroft *et al.* Govt. of India Press. 1959. 82 pp. No Price. A committee set up by the Ministry of Transport and Communications of India to determine *inter alia* "a formula for working out the "standard costs" of operation on the basis of which the losses of the Corporation might be subsidised by the Central Government."

Report on the I.C.A.O. Special Meeting, Montreal, February 1959. Decca Navigator Co. Ltd. 1959. 15 pp. No price. Compiled by observers from the company after the Special Meeting at which D.M.E.T. was recommended as the I.C.A.O. standard navigational aid, rather than the Decca system.

Scientific Research and Development in Colleges and Universities: Expenditures and Manpower 1953-4. National Science Foundation. U.S.G.P.O. 1959. 173 pp. 9s. (H.M.S.O.) The first full survey of its kind "it represents an initial effort in methodology and may form the pattern for future surveys." A similar study for 1957-58 is being prepared. There are 62 Tables and six charts.

Theoretical Elasticity. Carl E. Pearson. Harvard University Press (O.U.P.) 1959. 218 pp. Diagrams. 48s. *To be reviewed.*

Reports

AERODYNAMICS

Boundary Layer

Area-suction boundary-layer control as applied to the trailing-edge flaps of a 35° swept-wing airplane. W. L. Cook et al. *N.A.C.A. Report* 1370. 1958.
Three-component force data and pressure data presented show the effects of area suction boundary layer control on the longitudinal characteristics of a 35° swept-wing aeroplane. Also presented are pilots' opinions of the effects of boundary layer control on the handling qualities of the aeroplane.—(1.1.5.1 × 1.3.4 × 1.8.2.2).

Compressible laminar boundary layer over a yawed infinite cylinder with heat transfer and arbitrary Prandtl number. E. Reshotko and I. E. Beckwith. N.A.C.A. Report 1379 (*Supersedes N.A.C.A. T.N.* 3986). 1958.
Numerical solutions to the equations for the compressible laminar boundary layer have been obtained for the flow in the stagnation region of an infinite cylinder in yaw for large free stream Mach numbers and large yaw angles. The wall temperature was varied from zero to twice the stream stagnation temperature. Solutions are given for Prandtl numbers of 1 and 0·7.—(1.1.1 × 1.9.1).

Charts and tables for estimating the stability of the compressible laminar boundary layer with heat transfer and arbitrary pressure gradient. N. Tetervin. N.A.S.A. Memo 5-4-59L. *T.I.L.* 6398. *May* 1959.
The minimum critical Reynolds numbers for the similar solutions of the compressible laminar boundary layer computed by Cohen and Reshotko and also for the Falkner and Skan solutions as recomputed by Smith have been calculated by Lin's rapid approximate method.—(1.1.1 × 1.9.1).

A simple method for determining heat transfer, skin friction, and boundary layer thickness for hypersonic laminar boundary layer flows in a pressure gradient. M. H. Bertram and W. V. Feller. N.A.S.A. Memo 5-24-59L. *T.I.L.* 6408. *June* 1959.
A procedure based on the method of similar solutions is presented by which the boundary layer effects in a laminar hypersonic flow may be rapidly evaluated if the pressure distribution is known. The theory has been compared with results from heat transfer experiments on blunt-nose models at free stream Mach numbers of 4 and 6·8, including tests at angle of attack. Correlation of boundary layer thickness has also been shown.—(1.1.1.4 × 1.9.1).

Boundary layer transition on a 10-degree cone in the N.A.E. 30 × 16-inch wind tunnel. J. A. van der Blick. N.A.E. Report LR-232. Nov. 1958.—(1.1.2.4 × 1.12.1.3).

Compressible Flow *see also* Fluid Dynamics
Stability and Control
Wings and Aerofoils

Formulas pertinent to the calculation of flow-field effects at supersonic speeds due to wing thickness. K. Margolis and M. H. Elliott. N.A.S.A. Memo 4-3-59L. *T.I.L.* 6378. *May* 1959.
Expressions based on linearised supersonic flow theory are derived for the perturbation velocity potential in space due to wing thickness for rectangular wings with biconvex aerofoil sections and for various swept-back wings with wedge-type aerofoil sections. The complete range of supersonic speeds is considered subject to a minor aspect ratio restriction for the rectangular plan form and to the condition that the trailing edge is supersonic for the swept-back wings.—(1.2.3.1 × 1.10.1.2 × 1.6.1).

Supersonic and moment-of-area rules combined for rapid zero-lift wave-drag calculations. L. L. Levy. N.A.S.A. Memo 4-19-59A. *T.I.L.* 6392. *June* 1959.
The concepts of the supersonic area rule and the moment of area rule are combined to develop a method for calculating zero lift wave drag which is amenable to the use of ordinary desk calculators.—(1.2.2.1 × 25 × 22).

Controls *see* Boundary Layer

Fluid Dynamics

Fluid-dynamic properties of some simple sharp- and blunt-nosed shapes at Mach numbers from 16 *to* 24 *in helium flow. A. Henderson and P. J. Johnston. N.A.S.A. Memo* 5-8-59L. *T.I.L.* 6407. *June* 1959.
Flat plates, 5° and 10° wedges, and 5° and 10° cones were investigated. The flat plate results are for a leading edge Reynolds number range of 548 to 19,500.—(1.4 × 1.5.1.4 × 1.2.3 × 1.10.2.1).

Internal Flow *see also* Fluid Dynamics

Experimental investigation of expanded duct sections and screens for reducing flow distortions at subsonic flows. B. G. Chiccine and K. L. Abdalla. N.A.S.A. Memo 1-9-59E. *T.I.L.* 6382. *May* 1959.
Parameters investigated were expansion angle and length, area ratio, location of expanded section relative to engine face, and use of screens over a simulated engine-face Mach number range from 0·19 to 0·67.—(1.5.1.3).

Loads *see also* Compressible Flow
Stability and Control
Wings and Aerofoils
Aircraft Operation

A correlation of results of a flight investigation with results of an analytical study of effects of wing flexibility on wing strains due to gusts. C. C. Shufflebarger et al. *N.A.C.A. Report* 1365. 1958.—(1.6.3×2×24×33.1.2).

Modified matrix method for calculating steady-state span loading on flexible wings in subsonic flight. P. A. Gainer and W. S. Aiken. *N.A.S.A. Memo* 5-26-59L. *T.I.L.* 6409. *June* 1959.
The procedures comprise the solution of the equations of equilibrium by use of a matrix-iteration method, a simple check on the accuracy of the solution, and a method for obtaining the aerodynamic-induction matrix from previously published tables of downwash functions. The necessary parts of the downwash tables are included. The procedures are illustrated by a sample solution.—(1.6.1×2×1.10.1.2).

PERFORMANCE ESTIMATION *see* AIRCRAFT OPERATION

STABILITY AND CONTROL *see* BOUNDARY LAYER
DESIGN AND CONSTRUCTION
FLIGHT TESTING
MATHEMATICS
MISSILES

An analysis of the effects of aeroelasticity on static longitudinal stability and control of a swept-wing airplane. R. B. Skoog. *N.A.C.A. Report* 1298 (*Supersedes R.M.* A51C19). 1957.
A theoretical analysis of the effects of aeroelasticity on the stick-fixed static longitudinal stability and elevator angle required for balance of an aeroplane is presented together with calculated effects for a swept-wing bomber of relatively high flexibility.—(1.8.2.1×2).

Flight measurements of the effect of a controllable thrust reverser on the flight characteristics of a single-engine jet airplane. S. B. Anderson et al. *N.A.S.A. Memo* 4-26-59A. *T.I.L.* 6373. *May* 1959.—(1.8.2.1×13×2×27.1×5.3).

Tables for the rapid estimation of downwash and sidewash behind wings performing various motions at supersonic speeds. P. J. Bobbitt. *N.A.S.A. Memo* 2-20-59L. *T.I.L.* 6376. *May* 1959.
Tabulations are given of the downwash and sidewash due to a unit-strength horseshoe vortex located at 20 equally spaced positions along lifting lines of various sweeps. The charts are given for the purpose of estimating the spanwise variations of the flow field velocities and give longitudinal variations of the downwash and sidewash at a number of vertical and spanwise locations due to a unit-strength unswept horseshoe vortex. Illustrative computations are presented to demonstrate the use of the tables and charts and the rapidity with which answers can be obtained.—(1.8.0.2×1.10.1.2×1.6.1×1.2.3).

THERMOAERODYNAMICS *see* BOUNDARY LAYER
TESTING AND INSTRUMENTS

WINGS AND AEROFOILS *see also* COMPRESSIBLE FLOW
FLUID DYNAMICS
LOADS
STABILITY AND CONTROL
AEROELASTICITY
HYDRODYNAMICS

Surface pressure distribution at hypersonic speeds for blunt delta wings at angle of attack. M. O. Creager. *N.A.S.A. Memo* 5-12-59A. *T.I.L.* 6374. *May* 1959.
Surface pressures were measured on a blunt 60° delta wing at angles of attack from −10° to +10° for a Mach number of 5·7 and a Reynolds number per inch of 20,000. These surface pressures and pressures reported elsewhere for similar test bodies at Mach numbers of 11·5 in air and 13·3 in helium are compared with those predicted by a method developed previously for two-dimensional flow.—(1.10.2.2×1.6.1×1.2.3).

Basic pressure measurements at transonic speeds on a thin 45° sweptback highly tapered wing with systematic spanwise twist variations: untwisted wing. J. P. Mugler. *N.A.S.A. Memo* 10-20-58L. *T.I.L.* 6405. *Dec.* 1958.
Data are presented which were obtained in the Langley 8 ft. transonic pressure tunnel at Mach numbers from 0·800 to 1·200 through an angle of attack range from −4° to 12°. The wing has a taper ratio of 0·15, an aspect ratio of 4·0, is cambered, and has a thickened root section. Data were taken at stagnation pressures of both 1·0 and 0·5 atmosphere.—(1.10.2.2×1.6.1×2).

Transonic aerodynamic characteristics of two wedge airfoil sections including unsteady flow studies. P. J. Johnston. *N.A.S.A. Memo* 4-30-59L. *T.I.L.* 6406. *June* 1959.
A wind tunnel investigation has been conducted to determine the two-dimensional aerodynamic characteristics of a single 20 per cent thick wedge aerofoil section. The results are compared with those of a diamond profile having the same leading edge included angle. Pulsating pressure measurements are presented for the single wedge for leading edge radii of 0, 0·5, and 1 per cent of the chord.—(1.10.2.1×1.2.2).

Drag measurements on AGARD model 'A' in the N.A.E. 30×16-inch wind tunnel and comparison with other data. J. A. van der Bliek. *N.A.E. Report LR*-233. *Nov.* 1958.
The zero lift drag was measured on the AGARD standard model "A" at Mach numbers from 0·4 to 2·0 and at Reynolds numbers from 2·5 to 4·8×10⁶. Both the full model and the half model techniques were used. Comparisons were made with other experimental results.—(1.10.1).

A method for solving the subsonic problem of the oscillating finite wing with the aid of high-speed digital computers. V. J. E. Stark. *SAAB T.N.* 41. *Dec.* 1958.—(1.10.1.2×2×1.6.3).

HELICOPTER AERODYNAMICS

Flight investigation of effects of transition, landing approaches, partial-power vertical descents, and droop-stop pounding on the bending and torsional moments encountered by a helicopter rotor blade. L. H. Ludi. *N.A.S.A. Memo* 5-7-59L. *T.I.L.* 6381. *May* 1959.—(1.11.1).

TESTING AND INSTRUMENTS *see also* BOUNDARY LAYER
POWER PLANTS

Some notes on the use of resistance thermometers for the measurement of heat transfer rates in shock tubes. B. D. Henshall and D. L. Schultz. *C.P.* 408. 1959.
A brief description is given of the model calibration technique and the experimental data reduction used at the N.P.L. in the determination of heat transfer rates in a hypersonic shock tube. Estimates are made of the probable accuracy of such measurements, and specimen calculations and results are given.—(1.12.6×1.9.1).

Some experiments with static tubes at transonic speeds in a slotted-wall wind tunnel. E. N. E. Rogers and I. M. Hall. *C.P.* 430. 1959.
Some results obtained with five static tubes at transonic speeds in an N.P.L. slotted wall tunnel (17 in.×14 in. working section) are discussed. The effect of model size is considered, and a comparison made with free-flight tests, for tubes having a hemispherical nose. A brief analysis is made of the flow development with stream Mach number and of the problems associated with shock wave reflection.—(1.12.1.2×1.12.5).

Notes on half model testing in wind tunnels. J. A. van der Bliek. *N.A.E. Report LR*-235. *Jan.* 1959.
A short review is given of the effects associated with half model technique in wind tunnel testing. These include tunnel wall boundary layer and gap between model and tunnel wall. Reflection plane configurations are considered. Some experimental data on comparison between full and half model tests are given.—(1.12.1).

AEROELASTICITY

See also AERODYNAMICS—LOADS
STABILITY AND CONTROL
WINGS AND AEROFOILS
PROPELLERS
STRUCTURES—LOADS

The aerodynamic effects of aspect ratio on control surface flutter. H. Hall and E. W. Chapple. *C.P.* 434. 1959.
A series of low speed flutter tests to obtain a direct measurement of the aerodynamic effects of aspect ratio on wing-span ailerons, the wings having root flexibilities in roll and pitch, are described. Provision was made for mass-balancing the ailerons. Some general conclusions are drawn concerning the effects of aspect ratio and mass-balance on control surface flutter.—(2×1.10.2.2).

Incompressible flutter characteristics of representative aircraft wings. C. H. Wilts. *N.A.C.A. Report* 1390. (*Supersedes N.A.C.A. T.N.* 3780). 1958.

The results of a detailed study of the flutter characteristics of four representative aircraft wings are given. During the course of this investigation, eight important parameters of each wing were varied and, in addition, the effects of mass, inertia, pitching spring, and location of a concentrated mass were investigated for all four wings and several sweepback angles.—(2×1.10.1.2).

AIRCRAFT

See AIRCRAFT OPERATION

DESIGN AND CONSTRUCTION

Sampled-data techniques applied to a digital controller for an altitude autopilot. S. F. Schmidt and E. V. Harper. N.A.S.A. Memo 4-14-59A. *T.I.L.* 6391. *June* 1959.
Two alternative design procedures are presented which result in satisfactory performance. Analog computer results are presented illustrating transient performance of the systems. Included, also, are some root-locus plots demonstrating the sensitivity problem of the finite settling time design.—(4×1.8.0.1).

AIRCRAFT OPERATION

See also AERODYNAMICS—STABILITY AND CONTROL
FLIGHT TESTING

The suitability of the Bendix flight director system for Australian operation. J. R. Baxter. ARL Tech. Memo HE.3. April 1959.
The operation of the Bendix Flight Director System on Australian V.A.R. ranges has raised certain problems in information presentation and pilot setting. Two arrangements already tried in Australia have been critically analysed and an alternative proposal is made.—(5).

Crash-fire protection system for T-56 turbopropeller engine using water as cooling and inerting agent. A. M. Busch and J. A. Campbell. N.A.S.A. Memo 6-12-59E. *T.I.L.* 6394. *May* 1959.
The system includes an engine fuel-manifold shut-off and drain system for rapidly extinguishing the normal combustor flame, and a water spray system for cooling and inerting the hot interior engine parts likely to ignite crash-spilled fuel.—(5×3×27.5).

Airplane measurements of atmospheric turbulence at altitudes between 20,000 *and* 55,000 *feet for four geographic areas. T. L. Coleman and M. T. Meadows. N.A.S.A. Memo* 4-17-59L. *T.I.L.* 6395. *June* 1959.
An analysis is presented of data during research flights covering approximately 150,000 miles over the western part of the United States, England and Western Europe, Turkey, and Japan. The gust experience for the four geographic areas is compared and the variations in the gust frequencies and the percentage of flight distance in rough air with altitude are indicated. The results on the variation of the percentage of flight distance in rough air with altitude are compared with previous estimates.—(5×1.6.3×24).

Flugmechanische Betrachtungen zur Verkürzung von Start-und Landestrecken. H. Herb. DFL-Bericht Nr 80. 1959. (In German).—(5×27×1.7).

Investigation of the characteristics of an acceleration-type take-off indicator in a large jet airplane. J. J. Kolnick and E. Rind. N.A.S.A. Memo 4-21-59L. *T.I.L.* 6379. *May* 1959.
An investigation was conducted to determine the characteristics of an acceleration-type take-off indicator in a large jet aeroplane to detect subnormal aeroplane acceleration during take-off such as occasioned by loss of engine thrust or abnormal aeroplane resistance.—(5.3×18.2).

Über die Einheit von Ausrüstung und Flugzeug. H. Koppe. DFL-Bericht Nr 111. 1959. (In German).—(5.3).

Full-scale investigation of several jet-engine noise-reduction nozzles. W. D. Coles and E. E. Callaghan. N.A.C.A. Report 1387 (*Supersedes N.A.C.A. T.N.* 3974). 1958.
A number of nozzles which use the mixing interference of adjacent jets for noise suppression were investigated. A method of calculating the limiting frequency affected by multiple-slot nozzles is presented.—(5.6×27.4).

FLIGHT TESTING

See also AERODYNAMICS—STABILITY AND CONTROL

The influence of drag characteristics on the choice of landing approach speeds. D. Lean and R. Eaton. C.P. 433. 1959.
A study has been made of the lift-drag characteristics of 19 jet aircraft which have minimum comfortable approach airspeeds that are fairly well established.—(13×5×1.8.0.1).

Measurement of static pressure on aircraft. W. Gracey. N.A.C.A. Report 1364 (*Supersedes T.N.* 4184). 1958.
Existing data on the measurement of static pressure by means of static pressure tubes and fuselage vents at subsonic, transonic, and supersonic speeds are presented. Static pressure errors are given for isolated tubes and vents and for installations ahead of the fuselage nose, wing tip, and vertical tail fin, and on the fuselage. Various methods of calibrating installations in flight are briefly discussed.—(13).

HYDRODYNAMICS

Hydrodynamic characteristics of two low-drag supercavitating hydrofoils. J. R. McGehee and V. E. Johnson. N.A.S.A. Memo 5-9-59L. *T.I.L.* 6399. *June* 1959.
Aspect ratio one and aspect ratio three hydrofoils were tested for a range of cavitation numbers from 0 to approximately 6. The aspect ratio one hydrofoil was also tested at zero cavitation number with two sets of end plates having depths of $\frac{1}{2}$ and $\frac{1}{4}$ chords. The experimental results for the hydrofoils without end plates were compared with a three-dimensional zero cavitation number theory.—(17×1.10.2.2).

INSTRUMENTS AND EQUIPMENT

See also AIRCRAFT OPERATION

On the principles and angular accuracy of monopulse radar. G. Hellgren. SAAB T.N. 42. *March* 1959.
The general principles of one-dimensional angle-measuring monopulse radar are discussed. An expression for detector output signal for one or several targets is derived, and the two limiting cases of monopulse radar, the amplitude-comparison and phase-comparison types are defined and studied in some detail.—(18.1).

MATERIALS

Boron and zirconium from crucible refractories in a complex heat-resistant alloy. R. F. Decker et al. *N.A.C.A. Report* 1392 (*Supersedes N.A.C.A. T.N.* 4049; *T.N.* 4286). 1958.
The effects of the presence of small amounts of boron and zirconium in a titanium and aluminium hardened nickel-based alloy were studied. Beneficial increases in high temperature properties of the alloy were obtained by melting in magnesia or zirconia crucibles.—(21.2.2).

Exploratory investigation of advanced-temperature nickel-base alloys. J. C. Freche and W. J. Waters. N.A.S.A. Memo 4-13-59E. *T.I.L.* 6385. *May* 1959
A series of nickel-base alloys that do not require vacuum melting techniques and that generally provide good stress rupture and impact properties was developed for possible aircraft turbine blade or space vehicle applications. The strongest composition displayed stress rupture lives of 384 and 574 hours in the as-cast and homogenised conditions, respectively, at 1800°F and 15,000 lb./in.². Several of the compositions evolved could not be broken in a low-capacity Izod impact tester.—(21.2.2).

MATHEMATICS

See also AERODYNAMICS—COMPRESSIBLE FLOW

Investigation of a nonlinear control system. I. Flugge-Lotz et al. *N.A.C.A. Report* 1391. (*Supersedes N.A.C.A. T.N.* 3286). 1958.—(22×1.8.0.1).

Some general concepts of dynamics and their application to the restricted three-body problem. P. A. Lagerstrom and M. E. Graham. Douglas Report SM-23401. *Dec.* 1958.
Kinematics is reviewed with special attention to co-ordinate transformations. Conservation theorems and Lagrange's and Hamilton's equations of motions are discussed and illustrated by simple examples. Integrals of motion of the two-body problem are found and with their aid the two-body problem

is reduced to two uncoupled central-force problems. The general three-body problem is formulated and integrals of motion found.—(22.2).

METEOROLOGY

See also AERODYNAMICS—LOADS
AIRCRAFT OPERATION

Atmospheric turbulence encountered by Super Constellation aircraft. J. R. Heath-Smith. C.P. 432. 1959.
Counting accelerometer records representing one million miles were obtained from Super Constellation aircraft flying between Australia and the U.K. and across the Pacific Ocean and Indian Ocean.—(24).

MISSILES

See also AERODYNAMICS—COMPRESSIBLE FLOW

Analytical investigation of a flicker-type roll control for a Mach number 6 missile with aerodynamic controls over an altitude range of 82,000 *to* 282,000 *feet. R. R. Lundstrom and R. I. Whitman. N.A.S.A. Memo* 4-23-59L. *T.I.L.* 6397. *May* 1959.—(25.1×1.8.1.1).

POWER PLANTS

See also AERODYNAMICS—STABILITY AND CONTROL
AIRCRAFT OPERATION
STRUCTURES—LOADS

*Full-scale propulsion testing in wind tunnels. R. W. Hensel and H. K. Matt. Astia AD-*214888. *May* 1959.
A historical review of the development of full-scale propulsion testing in wind tunnels is presented, covering the range from early low subsonic speeds at close to sea level conditions to high supersonic speeds at altitudes of over 100,000 ft. The main emphasis is on the higher velocity, continuous flow wind tunnels from transonic upward. The various techniques for establishing matched altitude conditions of Mach number, pressure, and temperature are discussed. Detailed descriptions are given of particular items peculiar to propulsion testing in wind tunnels. Finally, future trends in propulsion testing in winds tunnels are postulated with emphasis on rocket installation testing at high altitudes.—(27×1.12.1).

Exploratory investigation of aerodynamic flameholders for afterburner application. H. F. Butze and A. J. Metzler. N.A.S.A. Memo 4-9-59E. *T.I.L.* 6384. *May* 1959.
The flame-holding capabilities of a number of aerodynamic jets and of combinations of these jets with a V-gutter flame holder were investigated at severe afterburner operating conditions.—(27.1×27.4).

Evaluation of transpiration-cooled turbine blades with shells of "Poroloy" wire cloth. H. T. Richards. N.A.S.A. Memo 1-29-59E. *T.I.L.* 6383. *May* 1959.
Permeability of aerofoil shell and soundness of structure were examined on a group of strut-supported, transpiration cooled turbine blades. Four were used to evaluate durability in a turbo-jet engine at an average turbine inlet temperature of approximately 1670°F and a turbine tip speed of approximately 1305 ft./sec.—(27.1).

Untersuchung der Spaltverluste von Turbinenletträdern mit bewegter Spaltwand. J. Rehbach. DFL-Bericht Nr. 90. 1959. (In German).—(27.1).

Experimental altitude performance of JP-4 fuel and liquid-oxygen rocket engine with an area ratio of 48. *A. Fortini* et al. *N.A.S.A. Memo* 5-14-59E. *T.I.L.* 6393. *May* 1959.
The performance of a rocket engine having a nozzle area ratio of 48 was experimentally measured at four altitudes and corrected to vacuum conditions. A comparison of experimental performance with that of a sea-level engine having an area ratio of 5·5 was made. Altitudes were obtained by an ejector system utilising the rocket exhaust gas as the pumping fluid. The results also include measured heat transfer rates and heat loads of the engine.—(27.3).

PROPELLERS

A wind-tunnel investigation of the aerodynamic characteristics of a full-scale supersonic-type three-blade propeller at Mach numbers to 0·96. *A. J. Evans and G. Liner. N.A.C.A. Report* 1375 (*Supersedes RM* L53F01). 1958.
The investigation was made in the Langley 16 ft. transonic tunnel with a 6000 h.p. propeller dynamometer. The tests covered a range of blade angles from 20·2° to 60·2° at forward Mach numbers up to 0·96. The effects of compressibility on the aerodynamic characteristics of the propeller and comparisons of the experimental results with calculated results are presented.—(29.7).

A wind-tunnel investigation of the stall-flutter characteristics of a supersonic-type propeller at positive and negative thrust. V. L. Rogallo and P. F. Yaggy. N.A.S.A. Memo 3-9-59A. *T.I.L.* 6377. *May* 1959.
Propulsive thrust and power characteristics were obtained in addition to thrust and torque distributions. The inflow velocity was also measured. Flutter characteristics were obtained at both positive and negative thrust conditions and with the thrust axis inclined. A correlation is made of the flutter stresses and aerodynamic section thrust characteristics.—(29×2×33.2.4.1.4).

Flight performance of a transonic turbine-driven propeller designed for minimum noise. T. C. O'Bryan and J. B. Hammack. N.A.S.A. 4-19-59L. *T.I.L.* 6396. *May* 1959.—(29.1×29.6×29.7).

STRUCTURES

LOADS *see also* AERODYNAMICS—LOADS

Calculated and measured stresses in simple panels subject to intense random acoustic loading including the near noise field of a turbojet engine. L. W. Lassiter and R. W. Hess. N.A.C.A. Report 1367. 1958.
Flat 2024-T3 aluminium panels were tested in the near noise fields of a 4 in. air jet and turbo-jet engine. The measured stresses are compared with those calculated by generalised harmonic analysis and are found to be in good agreement.—(33.1.2×33.2.4.5.10×27.1).

Experimental influence coefficients and vibration modes of a multispar 60° *delta wing. D. E. Weidman and E. E. Kordes. N.A.S.A. Memo* 2-4-59L. *T.I.L.* 6375. *May* 1959.
Test results are presented for both symmetrical and antisymmetrical static loading of a wing model mounted on a three-point support system. The first six free-free vibration modes were determined experimentally. A comparison is made of the symmetrical nodal patterns and frequencies calculated from the experimental influence coefficients.—(33.1.2×33.2.3.2.10×2).

THEORY AND ANALYSIS *see also* PROPELLERS
STRUCTURES—LOADS
TESTING

Investigation of the structural behavior and maximum bending strength of six multiweb beams with three types of web. J. P. Peterson and W. E. Bruce. N.A.S.A. Memo 5-3-59L. *T.I.L.* 6380. *May* 1959.
The results of bending tests on six multiweb beams of optimum weight-strength design are presented. The internal structure of the beams consisted of various combinations of two types of full-depth solid webs and a post-stringer web. The observed structural behaviour, buckling load, and failing load of the beams are compared with results obtained by the use of existing methods of analysis and found to be quite predictable.—(33.2.4.1.2×33.4).

*Experiments on the post buckling behaviour of simply supported panels that change in thickness across the bay. J. P. Benthem. NLL-TM S.*522. *March* 1958.
On 16 panels that change their thickness discontinuously (on flat plates, strips are glued) compression tests were performed. These panels represent skin bays which are stabilised by longitudinal stiffeners, the stiffener flanges being bonded to the skin. Several symmetrical and unsymmetrical configurations with different thickness and width ratios were taken. Results are given as relations between mean (compressive) stress, edge stress, edge strain and strain at initial buckling. By proper distribution of material along the bay width a panel may gain in stiffness and ultimate load appreciably.—(33.2.4.6.1).

TESTING

Beitrag zur Ausarbeitung normungsfähiger Prüfverfahren für Metallklebeverbindungen. H. Meckelburg. DFL-Bericht Nr. 103. 1959.—(33.3.2×33.2.4.13).

WEIGHT ANALYSIS AND CONTROL *see* THEORY AND ANALYSIS

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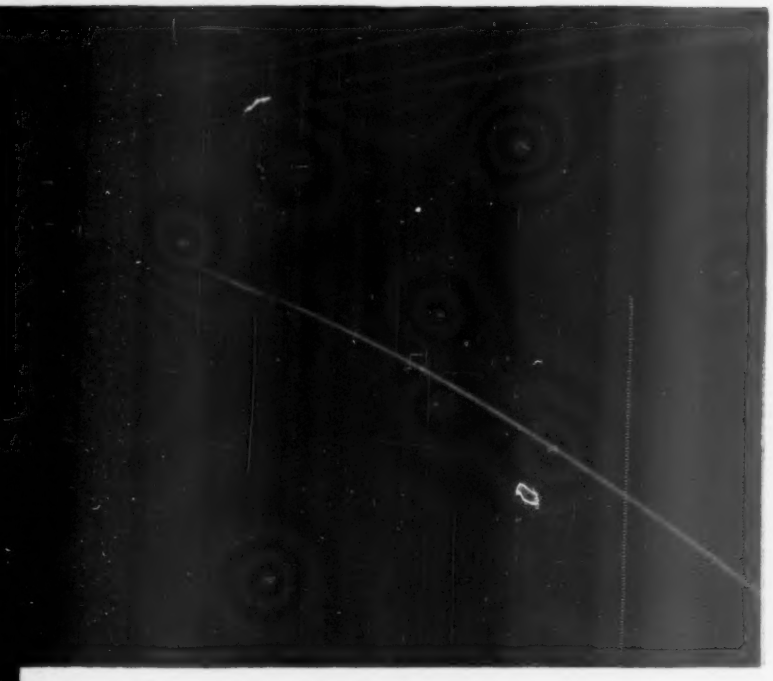

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